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THEORETICAL HEAT BALANCE STUDY OF A MANNED ORBITAL SPACE STATION

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SUMMARY

A comprehensive analytical procedure for the determination of the thermal balance of a manned orbital space station has been developed. The procedure includes a semi-empirical method for finding the view-factors for external radiation calculations over the vehicle surface. The procedure is applied to the erectable Manned Orbital Space Station currently under study at the NASA Langley Research Center.

The results of the analysis prove conclusively the feasibility of maintaining a "shirt-sleeve" temperature environment within the living quarters of the vehicle with a semi-active temperature control system. The design philosophy consists of providing external radiation characteristics which maintain comfortable internal air temperature during the "hot" portion of the orbit. The torus, which comprises the living area of the vehicle, is isolated from the life-support cylinder, which is allowed to run at higher temperatures during this part of the orbit. When the vehicle reaches the shadow portion of the orbit the cylinder air is introduced into the torus and allowed to mix with the torus air, minimizing the torus temperature fluctuation.

Off-design conditions are investigated to determine the fail-safe characteristics of this type of system.

INTRODUCTION

The concept of an erectable Manned Orbital Space Station as envisioned by NASA's Langley Research Center requires the simultaneous solution of several interesting and unique heat transfer problems (See Figure 1). The complex geometry, fixed axial orientation during orbit and the usual requirements of low weight and high reliability present severe design requirements. These are further complicated by the necessity to maintain a "shirt-sleeve" life support environment within the vehicle. The successful solution of these problems requires an analysis which allows each variable factor to be examined individually for its effects on the overall thermal balance of the vehicle. Such an analytical procedure has been devised and is presented herein. From the outset, the thermal balance system was limited to a semi-active design. In no case would a mechanical device or power consuming equipment be used to control the vehicle temperature. One concession was made in this regard, that of allowing the use of controlled internal air circulation to modulate the heat flux rates within the vehicle.

Major emphasis was placed on the establishment of methods to handle the various phases of the analysis rather than on the rapid determination of numerical values of temperature over the vehicle. The justification for this approach is two-fold. First, it has been demonstrated throughout the course of this work that the configuration of the vehicle is not firmly established. Any numerical results applicable to one configuration would not necessarily apply to others. Second, it would be highly impractical to attempt to cover all the possible permutations and combinations of surface coatings, orientation sequences, orbits, internal power dissipation schedules, and airflow rates and patterns even for one configuration. This is not to say that specific sets of design data have not been investigated in detail. Indeed, a series of coating characteristics have been compared for a given orbit and vehicle configuration. In addition, several off-design conditions have been analyzed to determine the "fail-safe" capability of the recommended thermal system. Basically, however, the aim of this study has been the establishment of an analytical procedure sufficiently complete and accurate so that it could be used to design a semi-active thermal control system for a manned orbiting vehicle of complex geometric shape.

The major sources of heat to the vehicle are the sun and the earth, by external radiation, and the internal power dissipation of electrical and electronic equipment by radiation, conduction and convection. Since most external surface areas on the vehicle receive radiation not only from the sources directly, but by reflected source radiation from other areas, a semi-empirical method for the determination of the total radiation absorbed by any surface area has been devised. The radiant interchange between interior surface areas is calculated by a procedure which takes advantage of the fact that a torus can be described analytically.

The problems of off-design conditions have been approached from the standpoint of providing maximum time for vehicle repair in the event of failure of some necessary vehicle component. Obviously, the desire to avoid an active temperature control system imposes certain restrictions on the capability of the system to handle all exigencies. Many times, however, emergency conditions are self-compensating. For example, a failure of the stabilization system could cause the solar collector to be oriented 180° away from the sun. This would expose the maximum area of the torus to direct solar radiation. On the other hand, the misorientation of the collector would produce a power failure which would eliminate the internal heat flux. This compensating effect prolongs the time in which repairs can be attempted before the vehicle would have to be abandoned.

The general philosophy which has been followed in the design of the thermal control system is based on producing a "cold-running" torus to which heat can be added as required. This extra heat, when needed, is drawn from the life support cylinder, which is allowed to run hotter than the torus. During the hot part of the orbit, the exchange of air between torus and cylinder is curtailed and the cylinder temperature rises. When the vehicle enters the earth's shadow and heat is required in the cylinder, the air in the cylinder is mixed with torus air to replace the heat lost from the torus. For the most part, emergencies result in a cold vehicle which it is felt is much easier tolerated and/or compensated for than one of excessive temperature.

SYMBOLS

$F(\eta)$	=	View factor as a function of orientation angle.
$FS(\eta)$	=	View factor of space.
H	=	Altitude of vehicle above earth's surface.
h	=	Film coefficient.
L	=	Distance from earth surface area to vehicle.
Q	=	Heat flux.
r	=	mean radius of earth.
T	=	Temperature °R
W	=	Solar constant = 440 BTU/HR-FT^2
α	=	Absorptivity
α	=	Central angle of spherical cap
β	=	Angle between outward normal to earth surface area and line from earth surface area to vehicle.
γ	=	Sweep angle of spherical cap
ϵ	=	Emissivity
ζ	=	Angle between earth-sun line and earth-to-vehicle line.
η	=	Angle between vehicle axis and line from external radiation source to vehicle.
θ	=	Position of vehicle in orbit measured from noon position.
ν	=	Rotational position angle of vehicle
ρ	=	Angle between earth-sun line and normal to earth surface area.
σ	=	Stephan-Boltzmann Constant = $.173 \times 10^{-8}$ BTU/HR-FT^2-$°F^4$

ϕ	=	Angle of inclination of orbit to earth-sun line.
A_L	=	Albedo

<u>Subscripts</u>

E	=	Earth
s	=	Solar
v	=	Vehicle
1	=	Source
2	=	Receiver

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DETAILS OF STUDY

A. General

There are several dominant characteristics of the Manned Orbital Space Station which affect its thermal balance. For one thing, the vehicle is designed to orbit the earth with its axis of symmetry fixed with relation to the sun. The solar collector, therefore, is the only component of the vehicle which receives direct solar radiation under normal conditions of operation. The remainder of the vehicle is subjected only to earth radiation, albedo, and radiation from the underside of the collector. The erectable structure is of relatively low thermal capacity which implies a marked tendency for the vehicle temperatures to follow closely the variable flux rates during orbit.

These flux rate variations are the result of two factors. First, the orientation of the vehicle with respect to the earth changes with position in orbit. Second, the vehicle generally enters the shadow of the earth and receives no solar radiation or albedo during that period. Both these variations can be eliminated if the angle of inclination of the orbital plane with respect to the earth-sun line is 90°. In that orbit, the vehicle receives a minimum of albedo radiation, the amount being constant during the entire orbit (See Figure 18). The earth radiates at a constant rate to the vehicle since it "sees" the same view of the vehicle during the entire orbit.

A more reasonable selection of orbital inclination, based on purely physical considerations, would be made by launching easterly from Cape Canaveral. This would result in an inclination angle with respect to the ecliptic of anything between 0° and 55°, depending upon the date and time of launch. In general, the greater the inclination angle, the less extreme the fluctuation in incident radiation flux during the orbit. At the same time, the precession rate of the orbit decreases with increase in angle of inclination. In order to minimize the effect of orbital precession, the date and time of launch should be selected so that the desired angle of inclination is reached as a maximum when the flight is half completed. For purposes of temperature computation, an angle of inclination of 30° to the earth-sun line was arbitrarily selected.

In order to reduce the control requirements of the thermal balance, the variations in external flux absorbed by the vehicle should be minimized. The above discussion of orbital selection describes one method by which this is accomplished.

Another and much more effective method of controlling the flux rates into and out of the vehicle is the judicious selection of surface coatings. The influence of α_s & ϵ, the absorptivity to solar radiation and infrared emissivity, respectively, on the vehicle temperatures during orbit is considerable. By making proper use of these characteristics, the mean temperature and the extremes can be adjusted. All other things being equal, the higher the value of α_s/ϵ , the higher the mean temperature of the surface.

The analysis described in this report is based upon the following general design philosophy. The orbital inclination should minimize as much as possible the fluctuations in external flux during the orbit. The coatings which are applied should be selected so that the temperature of the air in the torus does not exceed the maximum allowable value at the hottest point in the orbit. This implies that the vehicle never needs cooling. On the other hand, inherent in this type of approach is the tendency of the vehicle temperature to drop below the allowable minimum during the cold portion of the orbit. In order to replace the heat which the torus tends to lose when in shadow, some source of heat must be provided which can be brought into the torus when needed. The life support cylinder and the equipment within it can be made to serve this purpose most adequately. The equipment within the cylinder has a higher maximum allowable temperature than the inside of the torus, and the cylinder coatings are selected to prevent the cylinder from exceeding that temperature at the hottest point in the orbit. During the hot portion of the orbit the heat from the equipment is transferred to the cylinder walls and face by conduction, connection, and radiation. The air in the cylinder is prevented from entering the torus during this portion of the orbit. When the torus temperature begins to drop due to reduced external flux, the equipment heat from the life support cylinder is brought into the torus by convection and mass transfer to compensate in part for the reduction in absorbed flux. The effect of this added heat is a minimized temperature variation in the torus.

In addition to introducing added heat into the torus during the cold portion of the orbit, it is possible to reduce the heat transfer rate out of the torus by reducing the velocity of the air adjacent to the walls. In the life support cylinder, the convection rate from the air to the walls should be similarly reduced. On the other hand, the rate at which heat is convected from the equipment to the air in the cylinder should be kept as high as possible.

B. Methods

1. View Factors for External Source

In order to compute the amount of incident radiation from external sources which is absorbed by the various surfaces of the satellite it is necessary to determine an appropriate set of view factors for the given configuration. The view factor is defined as the ratio of total radiation incident on a surface to the total radiation a flat plate of equal area would receive if it were oriented normal to the incident ray. These view factors would be, in general, a function of the two angles necessary to describe the orientation of the surface with respect to the source (See Figure 2). For a flat plate with no reflecting areas near it, these view factors are obviously determined by Lambert's Cosine Law. Likewise, view factors for a surface which can be approximated by several plane surfaces may be determined by applying Lambert's Law to each surface and taking a weighted average based on areas. In this case, however, care must be taken to account for the possibility of one surface casting a shadow on another, or more difficult, surfaces reflecting the radiation among themselves. When the surface is as highly complex as the cylinder-torus-collector configuration, analytical determination of these view factors is out of the question for any portion of the surface except sections which do not "see" other parts of the vehicle. Under these circumstances, an experimental method must be used to determine view factors.

The experiment must determine the percentage of full intensity normal radiation incident on each section of the vehicle from a point source at a great distance. For this purpose, a scale model of the vehicle was constructed.

Light was used as the radiation since it may be easily measured with solar cells or other simple detectors. Because the light is to simulate a point source at a great distance it must be placed at a sufficient distance from the model so that the incident energy is collimated. Generally, twenty or more model diameters is sufficient. The following procedures were followed. The test area was darkened to eliminate stray light to the model by building a cage around it. The inside surfaces of the cage were treated to prevent reflection.

A carbon-arc spotlight was used to provide high intensity light over the entire model (See Figure A). While the uniformity of the illuminated area was satisfactory, the arc had a tendency to flicker in intensity due to changes in arc gap. An attempt was made to use a tungsten filament lamp of high intensity. The problem of flicker was eliminated but the field was no longer uniform due to low-quality optics in the system. The tests were completed with the carbon-arc source.

The geometrically scaled model (1/12 scale) was instrumented with solar cells (See Figure C) at significant surface areas. A control cell was placed near the model normal to the incident light to act as an intensity monitor. The cells were calibrated prior to installation to insure that their response was linear with intensity variations. The model was first oriented with its axis colinear with the incident radiation. In this position it was rotated 360° about the axis and readings were taken every 30°. The axis was then set at an angle of 20° to the light beam and the readings were taken again for rotated positions through 360°. The tests were repeated every 20° of axis angle through 140°. (At that point the collector shaded almost the entire vehicle). The entire process was repeated for several coatings, diffuse and specular. The results are presented in Figures through . It should be noted here that the data is presented as a function of the vehicle axis angle only. The values with respect to rotation angle have been averaged out because the vehicle rotates during orbit. To investigate a non-rotating vehicle, it would be necessary to use each view factor as initially determined (that is, as a function of axis angle and rotational orientation).

In order to calculate the amount of earth radiation absorbed by a given portion of the satellite at any position in orbit the following procedure is used. The "visible" spherical cap of the earth is divided into a series of rings, the ring areas then are subdivided into segments (See Figure 2). The values of view factors over the vehicle for different axis angles are tabulated. For each portion of radiating surface on the earth, the angle between the vehicle axis and the line between radiating surface and vehicle are computed. The appropriate view factor for the given vehicle portion at that axis angle is determined from the table. This factor is called $\mathcal{F}(\eta)$. The radiation absorbed is then:

$$dQ_E = \frac{\sigma \epsilon T_E^4}{\pi} \left[\frac{\cos\beta \, \mathcal{F}(\eta) r^2 \sin\alpha \, d\gamma \, d\alpha}{L^2} \right]$$

This equation is then integrated over the entire spherical cap. For each new cap portion to be calculated a new value of η is computed and the appropriate value of $\mathcal{F}(\eta)$ is selected.

The amount of albedo radiation absorbed by a section of the vehicle is similarly calculated except that an added variable, the angle between the earth-sun line and the outward normal to any segment of the spherical cap is computed in order to determine the output of the cap surface. The equation to be integrated is as follows:

$$Q_{AL} = \frac{W \rho_E \alpha_s}{\pi} \int_{\alpha=0}^{\alpha_L} \int_{\gamma=0}^{2\pi} \frac{\cos\beta \, \cos\rho \, \mathcal{F}(\eta) r^2 \sin\alpha \, d\gamma \, d\alpha}{L^2}$$

The validity of the above described test procedure requires that the combination of reflectivity of the surface as measured and the test source spectrum be the same as that of the prototype and the radiation spectrum it will encounter. For example, if a diffuse grey surface is tested under visible light and the surface has a strong specular reflection component in the IR region of the spectrum, then the test values of $\mathcal{F}(\eta)$ cannot be used to calculate radiation from the earth. This of course results from the fact that the majority of the earth radiation intensity lies in the near-to middle-IR region of the spectrum. The albedo radiation calculation, on the other hand, can be safely made using the test values since albedo radiation falls predominantly in the visible light region.

Another factor to be avoided in scaling the model is the inadvertent construction of a black-body cavity on the model. If some portion of the prototype contains a fairly deep depression, it may begin to approach a black-body when reduced in size. In that event, the absorptivity of the coating which has been applied may not be the true absorptivity of that area. In the 1/12 scale model which was used, this problem did not arise.

In tests of the type described in this section, it is valid to interpolate between different coatings provided all are diffuse or equally specular. It is not correct to use view factors obtained for diffuse coatings to apply to specular coatings.

2. External Radiation

a. Solar Radiation

The amount of direct solar radiation which reaches any part of the vehicle is affected only by the configuration of the vehicle and does not change during orbit since the vehicle orientation is fixed with respect to the sun. The radiation itself is assumed to be completely collimated. Normally, the solar collector faces the sun and the remainder of the vehicle receives no direct solar radiation. In the investigation of misalignment of the vehicle, however, it is possible for parts of the torus and/or the life support cylinder to receive radiation from the sun. In that case, the above described view factors are used to determine the total amount of solar radiation absorbed by any surface area. The total solar radiation is merely the solar constant multiplied by the absorptivity of the surface to solar radiation and the appropriate view factor. If the orientation of the vehicle changes with respect to the sun during the orbit, the view factor for each area changes correspondingly. It is necessary to limit the solar radiation calculation to account for the satellite entering the shadow of the earth. If $\zeta > \pi/2$ and $|(H + r)(\sin \zeta)| \leq r$, then the satellite receives no solar radiation.

b. Earth Radiation

For purposes of calculation, the earth is assumed to be a black body radiator at 250°K. The amount of radiation reaching any surface area on the vehicle is determined by integrating the contribution of incremental areas on the "visible" spherical cap of the earth over the entire cap (See Figure 2). The angle η uniquely determines the view factor since the vehicle is assumed to be rotating about its axis of symmetry. The surface of the earth is treated as a Lambert's Law radiator. Thus, the intensity of radiation reaching the given satellite surface is the normal output of the earth's area times the cosine of the angle times the appropriate view factor divided by πL^2. The distance L is a function of the angle α, the angle between the satellite radius vector and a ring on the surface of the spherical cap.

$$L = \sqrt{2r\,(r + H)(1 - \cos\alpha) + H^2}$$

Angle β is also dependent only on the central angle α.

$$\beta = \alpha + \cos^{-1}\left[\frac{H + r(1 - \cos\alpha)}{L}\right]$$

The total area of the visible spherical cap is computed as follows:

$$A = \int_{\alpha=0}^{\alpha_L}\int_{\gamma=0}^{2\pi} r^2 \sin\alpha \; d\gamma \; d\alpha$$

where

$$\alpha_L = \cos^{-1}\frac{r}{H + r}$$

Then the total radiation from the earth is

$$Q_E = \frac{\sigma\alpha_V\, T_E^4}{\pi}\int_{\alpha=0}^{\alpha_L}\int_{\gamma=0}^{2\pi}\frac{\cos\beta\,\mathcal{F}(\eta)\,r^2\sin\alpha}{L^2}\,d\gamma\; d\alpha$$

The extent of the visible cap is limited by not allowing angle β to exceed $\pi/2$ during the integration process.

In the event that a condition of non-rotation of the satellite were to be investigated, another variable would be required to define the view factor for the various satellite areas. The angle ν can be used for this purpose, ν_o being some arbitrary location.

c. Albedo Radiation

The earth can be assumed to reflect solar radiation diffusely. The albedo of the earth is approximately 0.35. The amount of solar radiation reflected normal to any area on the earth's surface is, therefore, 0.35 times the solar constant times the cosine of angle ρ. How much of that is absorbed by any satellite surface area is calculated by multiplying by the appropriate form factor, the absorptivity to solar radiation and dividing by πL^2. Some areas of the earth which may be visible to the satellite lie in shadow and do not reflect solar radiation. If $\rho > \pi/2$, the area in question does not reflect. The same type of integration which applies for earth radiation is used in this case to determine the total radiation to each area on the satellite. The actual integrations described above are performed by a series of IBM 704 programs which were written suring the course of this study.

3. Radiation Exchange

The radiation interchange that takes place at any portion of the vehicle consists of incoming radiation from external sources, radiation out from the surface to space, and an exchange of radiation between surfaces, both by emission and reflection. The amount of radiation interchange which takes place at a given surface will vary with vehicle configuration and the reflectivity and emissivity of the surface coating, and must be determined for each such case. For a given configuration and coating, all the radiation emitted by a section of the vehicle eventually either escapes to space, is absorbed by another section of the vehicle, or is reabsorbed by the emitting surface. The percentage of radiation from any section falling into each of these categories will remain unchanged regardless of the intensity of radiation from the section. This is apparent from the following argument. Idealizing the vehicle as a large number of differential plane areas, there will be a finite number of paths that radiation from a certain section can take. A certain amount will escape directly to space. The remainder will impinge on different areas of the vehicle and either be absorbed or reflected. After each reflection a given amount will either escape to space, be reflected again or be absorbed by some section. Doubling the radiation leaving the original section will only double the amount of radiation following each of these paths. Thus the pattern of distribution of radiation from any given section is only a function of vehicle geometry and surface properties.

The amount of radiation escaping directly to space from any section may be accurately determined using the previously discussed view factors for radiation from an external source. Consider the vehicle to be surrounded by a sphere at the temperature of the section in question. Assume the sphere to be a black body. Determine the amount of radiation from this sphere that is absorbed by the section. This may be done by performing an integration over the sphere using the view factors for radiation from an external source, in the same manner that radiation from the earth's cap was determined. The amount of radiation emitted by the sphere and absorbed by the section is equal to the radiation emitted by the section and absorbed by the sphere. Since the total radiation from the surface can be calculated if the temperature and emissivity are known, the ratio of the radiation to the sphere to the total radiation can be computed. This is the view factor of space for the section.

In the case at hand, where the vehicle is rotating, the view factor of space for any section is only a function of η, the angle between a line from the vehicle to the external source and the vehicle axis of rotation. If $F_i(\eta)$ is the view factor for a section, S_i, then the view factor of space, FS_i, is given by

$$FS_i = 2 \int_0^{\pi} F_i(\eta) \sin\eta \, d\eta$$

The problem of experimentally determining the view factors from one vehicle section to another is more difficult since a test procedure analagous to that for radiation from an external source requires that each section act as a radiation source. It is possible, however, to resort to approximate analytical methods or estimates without loss of accuracy. In the vehicle under consideration only the view factors between the collector and the rest of the vehicle need be determined with any great degree of accuracy because the collector represents the only section of the vehicle with a high emissivity and with temperatures radically different from 70°F.

In computing view factors between sections, we are concerned with determining the view factors which describe how much radiation is absorbed by each surface. This is in contrast to <u>direct</u> view factors which merely describe how much radiation impinges on a surface without taking reflections into account. However, these direct view factors are more easy to determine since they are only a function of vehicle geometry and do not have to be recomputed for changes in the surface properties of the vehicle. The total view factors may be computed from the direct view factors merely by tracing radiation from surface to surface on the vehicle. Consider the radiation leaving one section and impinging directly on the remaining surfaces or escaping directly to space, according to the direct view factors for the surface. Each of the other surfaces absorbs some of the radiation and reflects the rest. The view factor of the reflecting surface determines the distribution of the reflected radiation. In this fashion the radiation is traced until all but a negligible amount is absorbed or escapes to space. This process can be programmed for a computer as a relatively simple matrix operation. One complicating factor is the possibility of strong specularity in the reflective properties of the surfaces. When a specular surface is being analyzed, it is necessary to divide the problem and treat the diffuse and specular components individually. The procedures are similar.

Internal view factors between surfaces within the vehicle are important because the inside of the vehicle will rely heavily on radiation exchange to aid in maintaining even temperature distributions over the surface. Generally, the internal geometry is more simple than external and will frequently yield to a numerical analysis. Such is the case with the torus. The direct view factors from any point on the surface to any other section of the surface are determined by taking a large number of rays from the point to all directions such that the differential areas associated with the rays cover all the region seen by the point. Each ray has a differential amount of radiation based on the cosine law and its differential area. By determining which section each ray intersects, and summing the radiation coming into each section from the rays that intersect it, the direct view factor may be determined. This process has been programmed for the IBM 704 Digital Computer. The results appear in Figure 21.

4. Total Vehicle Time History Evaluation

The heart of any transient temperature time history analysis for a vehicle of such complexity must be a large digital computer program. This program must be able to evaluate all modes of heat transfer between different sections and components of the vehicle and integrate flow rates to determine temperatures. Since one requirement of this vehicle is that temperatures should not fluctuate, greatly it is not necessary to use a complex numerical integration technique. Except for the case of the collector, which is handled independently anyway, temperature derivatives with time will change slowly and smoothly so that a simple predictor-corrector integration technique or even a first order prediction of the form

$$T_i (t + \Delta t) = T_i (t) + \dot{T}_i (t) \Delta t$$

will suffice. The temperature derivatives with time are merely the heat flow rates in or out of a section or component divided by the heat capacity of same. The bulk of the program is devoted to evaluating these flow rates.

The first step is to divide the vehicle into a number of isothermal sections and components. These sections should be small enough to make the assumption of constant temperature over a surface reasonable, and large enough so that an unwieldly number is not required. For an axially symmetric, rotating vehicle, these sections will be circumferential strips since radiation input about any circumference will be evened out by the rotation. For a non-rotating vehicle, further subdivision is necessary. In the preliminary design stage, equipment can only be described by weight and heat output. It must therefore be treated as a single component as a uniform temperature. Since the pattern of air circulation within the vehicle is one of the major factors to be determined, the internal vehicle air must be handled in a general fashion. This is best accomplished by assuming the vehicle to be subdivided internally into imaginary compartments such that each compartment contains a mass of air at a uniform temperature. Each of these masses of air will be considered as a component, having a fixed heat capacity. Circulation of air between these compartments will be discussed shortly. Once the subdivision into sections and components is accomplished, heat capacities and heat transfer properties of each section must be determined.

For a specific orbit, the following procedures would be followed:

1. Based on the tests to determine view factors for radiation from an external source the radiation input to each section from earth and albedo radiation, and possibly direct solar radiation is precomputed and put in table form as a function of orbital position and vehicle surface characteristics. Each skin section will then absorb external radiation as a function of its area and surface properties.

2. Each skin section will radiate externally as a function of its temperature and surface properties, and this

C. Solar Collector

The solar collector is a large disc which will be used either as a reflector for a solar boiler or as a mounting for a bank of solar cells.

The vehicle orientation is maintained so that the collector always faces the sun. While this is a great aid in shielding the remainder of the vehicle from direct solar radiation, the collector is subjected to an extreme variation in environment as the vehicle moves in and out of the shadow of the earth. A suggested basic collector design consists of a honeycomb aluminum disc with faces of .003" sheet and a .001" core, $\frac{1}{4}$" across the hex flats.

The major restriction on the collector heat balance will be maximum allowable temperature. A bonded honeycomb will restrict the maximum temperature to about 250°F while the use of solar cells will limit the temperature even more radically. Whether the collector is used as a reflector or a solar cell bank, it will have two significant heat transfer characteristics. It will be of low heat capacity and low conductivity. This means that a temperature time history of a point on the collector during orbit will show the temperatures to be no more than several degrees away from the equilibrium temperature at any time. Further, conduction between two different points on the collector will have a negligible effect on the temperatures. The low conductivity <u>through</u> the collector has an adverse effect. For the honeycomb construction mentioned, the conductivity is of the order of:

$$k = .8\ \Delta T + .028\ \Delta T^2\ \text{BTU/HR-FT}^2\text{-}^\circ\text{F}$$

where ΔT is the temperature differential between the two faces. This results in a temperature difference of 20 to 30 degrees Fahrenheit, an important factor when maximum temperatures approach a critical limit.

The conditions the collector encounters will be most strongly affected by the selection of a vehicle orbit. This will determine how long the vehicle is to be in sunlight. Because the heat capacity of the collector is so low, the peak temperatures are unaffected by orbit changes. Only the periods of time spent at those temperatures change.

There are two ways in which the temperature of the collector may be controlled. The major factor will be the selection of a coating for the torus side of the collector. It is assumed that there will be little choice in the sun side coating since that will be set by the function of the collector. If it is to be used as a reflector, the sun side will be polished aluminum. If solar cells are used, the cell characteristics will determine the absorption and radiation properties. The problem is to provide enough heat dissipation to keep the collector from overheating. This will result in high values of emissivity for the torus side. In this case, a second consideration becomes important; radiation interference from the torus. If a highly reflective surface is in close proximity to the collector, the effective radiating area will be reduced, raising the collector temperature.

An additional problem area is the center section of the collector behind the life support cylinder. This area can do no radiating from its back face and therefore must eliminate its heat from the sun side. This is accomplished by using a coating with a high value of ϵ / α_s such as a white porcelain enamel. The use of this area for solar cells is not recommended.

The analysis of the collector temperatures may be completed without the necessity of coupling the solution to the heat balance for the remainder of the vehicle. While it is necessary to know the temperatures of the collector throughout the orbit in order to perform the heat balance on the torus and the cylinder, the reverse is not true. Since the torus is to be maintained at a near constant temperature of 70°F the variation in radiation from the torus to the collector will not change significantly during an orbit. The following information is necessary to determine collector temperatures.

1. ϵ and α_s for the sun side and torus side.
2. Inclination of orbit to earth-sun line.
3. View factors to determine the amount of earth radiation and albedo to both sides of the collector.
4. View factors to determine the amount of radiation the collector receives from the torus and cylinder.

The determination and use of the view factors in 3 and 4 are thoroughly discussed in the section on Methods. These view factors must be determined for different positions on the collector since the geometry of the vehicle will result in a wide variation radially.

The sun side of the collector receives heat from the sun except when the vehicle enters the shadow of the earth. It is also subject to albedo and earth radiation for slightly more than half the orbit. It radiates to space at some sun side temperature, T_1, with a view factor of 100%. The portion of the absorbed heat not radiated away is conducted through the collector structure based on the conductivity and the temperature differential across the collector faces.

The torus side of the collector receives heat by conduction through the structure as stated above. It also receives earth radiation and albedo based on its position in orbit and the absorption characteristics of its surface. The torus is assumed to radiate to this side of the collector at some fixed temperature. The torus side of the collector radiates to space and to the torus at some temperature, T_2.

The amount of earth radiation and albedo reaching any area on the collector as a function of orbital position is determined as follows. For the sun side, the view factors for positions in space follow Lambert's Cosine Law. From a view factor standpoint, it is assumed that the collector behaves as though it were a flat plate.

On the torus side of the collector, the view factors of various angular positions in space are determined by the view factor tests described above. The view factors are used in the IBM 704 program for integrating radiation from the earth's and the minimum at $\theta = 180°$. The results show that the α_s/ϵ ratio is most significant with values of $\alpha_s/\epsilon < 1.0$ being most effective in preventing high temperatures. For a polished aluminum sun side, ($\alpha_s = 0.1$, $\epsilon = 0.05$), temperatures remain below 200°F when the torus surface is coated so that $\alpha_s \leq 0.4$ and $\alpha_s/\epsilon < 1.0$.

When the sun side has a higher absorptivity, such as the case of a collector covered with solar cells, it is necessary to conduct the heat through the collector and radiate it away on the torus side. In this case, values in the order of $\alpha_s = 0.4$ and $\epsilon = 0.8$ are required. In addition, it might be necessary to use a coating with low values of α_s/ϵ in a mosaic with the cells on the sun side. In general, the high temperature condition of the collector with solar cells can be relieved by moving the collector 7-8 feet from the torus. This effect is limited by the fact that the shielding quality of the collector is reduced when its distance from the torus increases.

For the area of the collector directly adjacent to the life support cylinder, it is imperative that the surface be able to radiate sufficiently from the top side to prevent excessive heat conduction into the cylinder. White porcelain enamel with $\alpha_s = 0.2$ and $\epsilon = 0.8$ would be a suitable coating.

spherical cap (earth radiation and albedo) to determine the total heat input. Solar radiation to the collector is the product of the solar constant, 440 BTU/HR FT^2, and α_s, the absorptivity of the collector surface to the solar spectrum. With this information the temperatures on the collector may be calculated for any location in the orbit by an iterative process as follows:

1. Determine the radiation input to the sun side.
2. Assume a value for T_1, the temperature on the sun side.
3. Calculate the radiation out of the sun side at temperature T_1. The difference between absorbed and reradiated flux is assumed to be conducted through the collector.
4. Compute the ΔT necessary to conduct the heat through, based on the known conductivity of the structure.
5. Solve for T_2, the temperature on the torus side of the collector, based on T_1 and ΔT.
6. Iterate on T_1, until a balance is obtained between radiation in and out at T_2 and conduction through the structure.

This procedure was applied to the collector at three radial positions for an orbit inclined at 30° to the earth-sun line using estimated view factors. More than 40 combinations of α_s and ϵ for the upper and lower surfaces of the collector were investigated using the IBM 704 collector temperature program. The results are presented in Figure 23. The maximum temperatures in all cases occured at $\theta = 0°$ (the noon position)

D. Torus and Life-Support Cylinder

The torus of the vehicle, which is used as the living quarters and work area for the crew must be maintained at a near constant temperature of 70°F. This requirement shapes the entire effort involved in the balance, that of minimizing the effects of the variations in incoming radiation to the vehicle during an orbit. In addition to maintaining the temperature of the air in the torus at a constant temperature the skin temperature should not change enough to cause discomfort to a person near a wall. The life-support cylinder houses the bulk of the equipment in the vehicle and must be maintained at temperatures which will prevent over-heating of the equipment. It should also be possible for the crew to enter the cylinder although not necessarily while all equipment is operating.

A vehicle in a 400 mile orbit, protected from directly incident sunlight by a collector, as this vehicle is, would tend to cycle in temperature about some mean temperature between 0°F and 50°F. The actual mean would depend on the α_s/ϵ ratio of the vehicle surface, with mean temperature increasing for increasing α_s/ϵ. With 10,000 BTU/HR being generated, appropriate α_s and ϵ values can be found to produce a mean temperature balance at 70°F. Figure 20 shows appropriate α_s/ϵ combinations for orbits of all inclinations. It should be pointed out that an aluminized mylar coating over the whole vehicle would result in excessive temperatures due to the inability of the skin to dissipate

Within the torus, the air should be introduced so that it flows at a high enough velocity around the walls to establish a film coefficient of 1-2 BTU/HR-FT^2-°F. While that velocity would be intolerable for the occupants, it can be produced by peripheral inlets near the floor. The air at the center of the torus cross-section would remain relatively still.

Since the solar collector depends to a large extent on its bottom surface (that facing the torus) to radiate sufficiently to prevent overheating of the honeycomb, it would be of advantage to move the collector sufficiently far from the torus so that the collector view factors to space are increased. This is especially desirable when the collector contains solar cells. In that case, the maximum allowable collector temperatures are lower than for a reflector configuration.

This report represents the completion and fulfillment of Phase "A" of contract No. NAS1-970 between Grumman Aircraft Engineering Corporation and NASA Langley Research Center.

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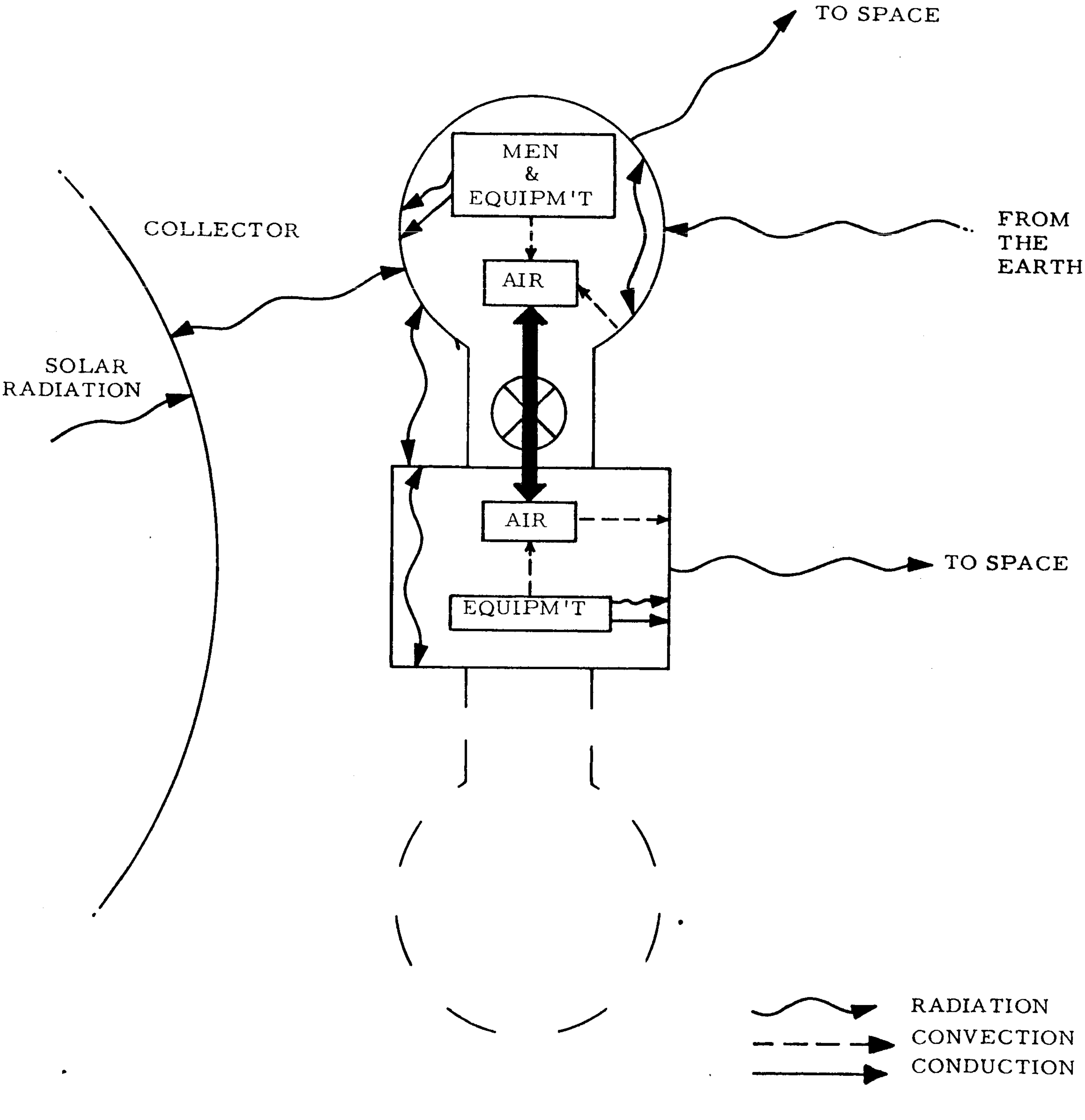

TO SPACE
MEN
&
EQUIPM'T
COLLECTOR
FROM
THE
EARTH
AIR
SOLAR
RADIATION
AIR
TO SPACE
EQUIPM'T
RADIATION
CONVECTION
CONDUCTION

Figure 1

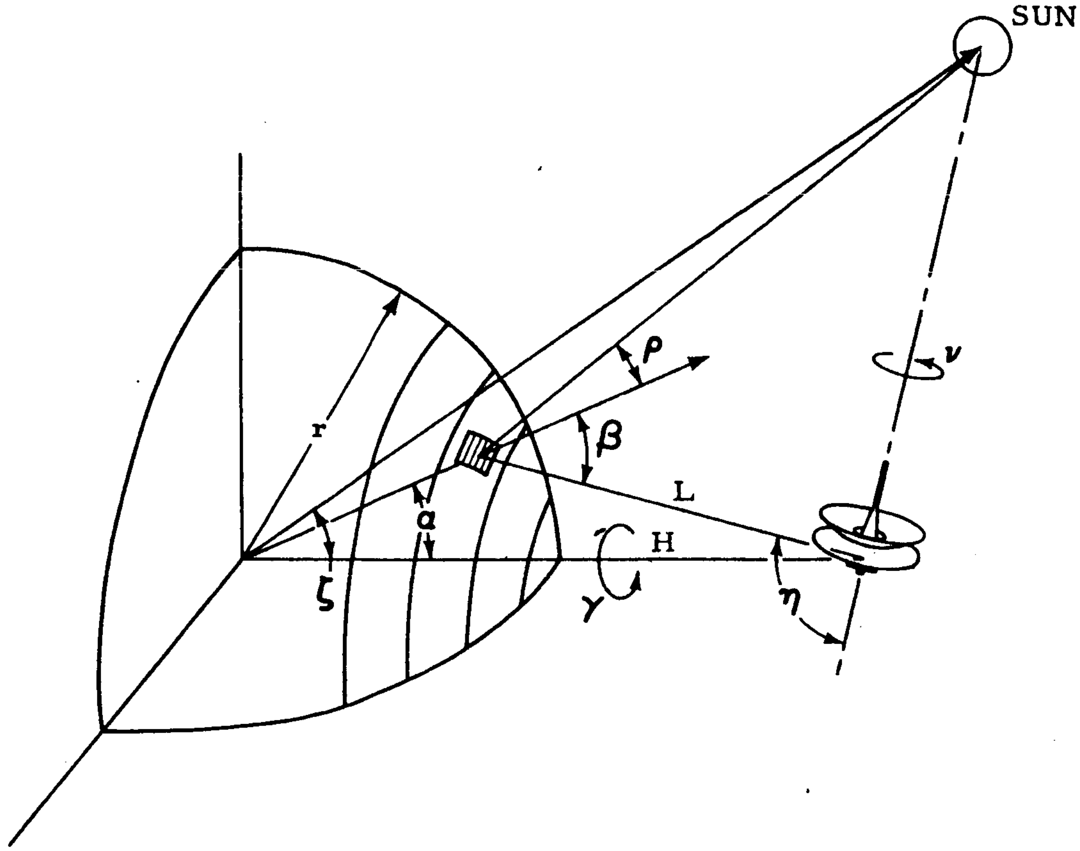


Figure 2

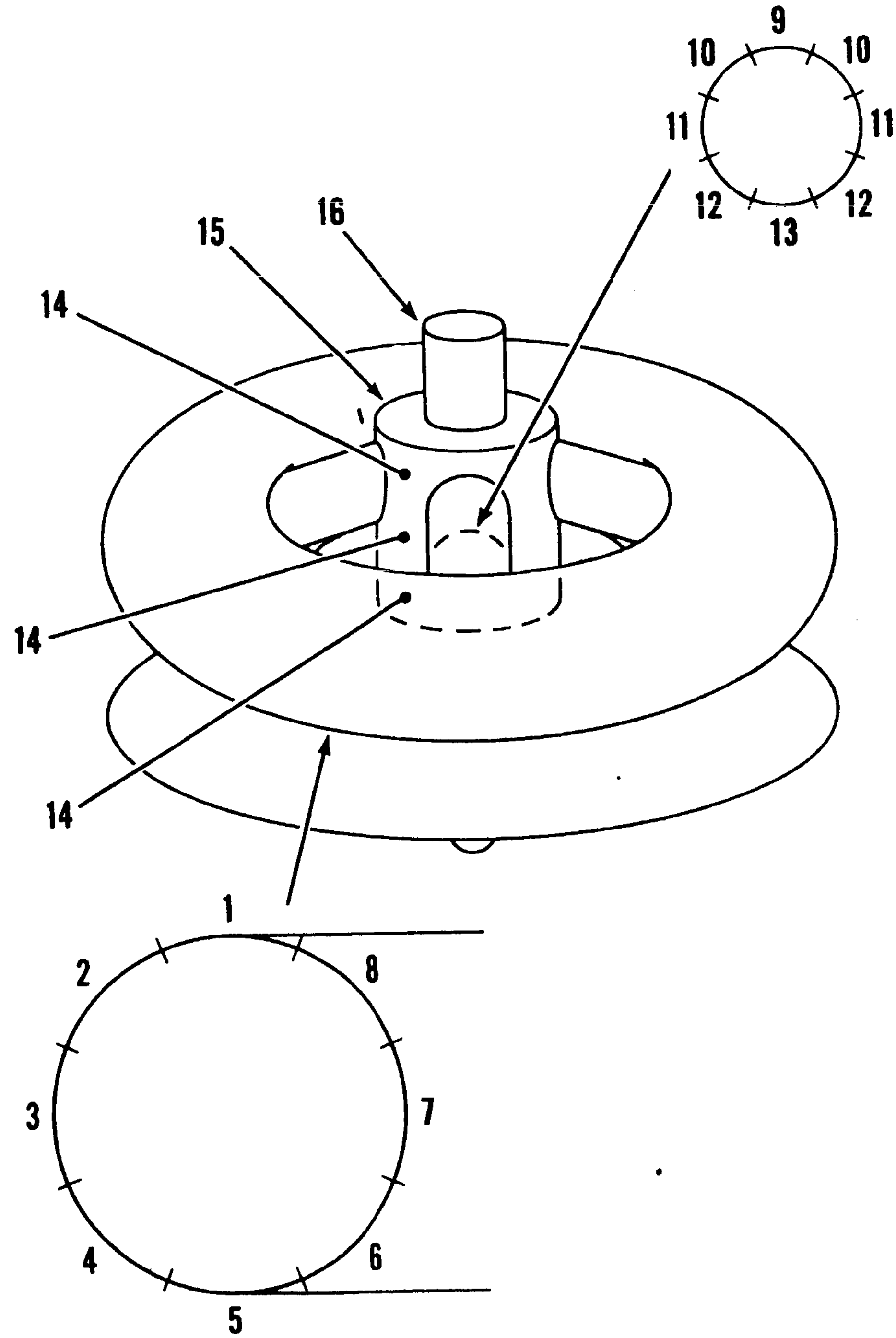


Figure 3

% OF FULL INTENSITY FOR FOUR TORUS POSITIONS - WHITE ENAMEL

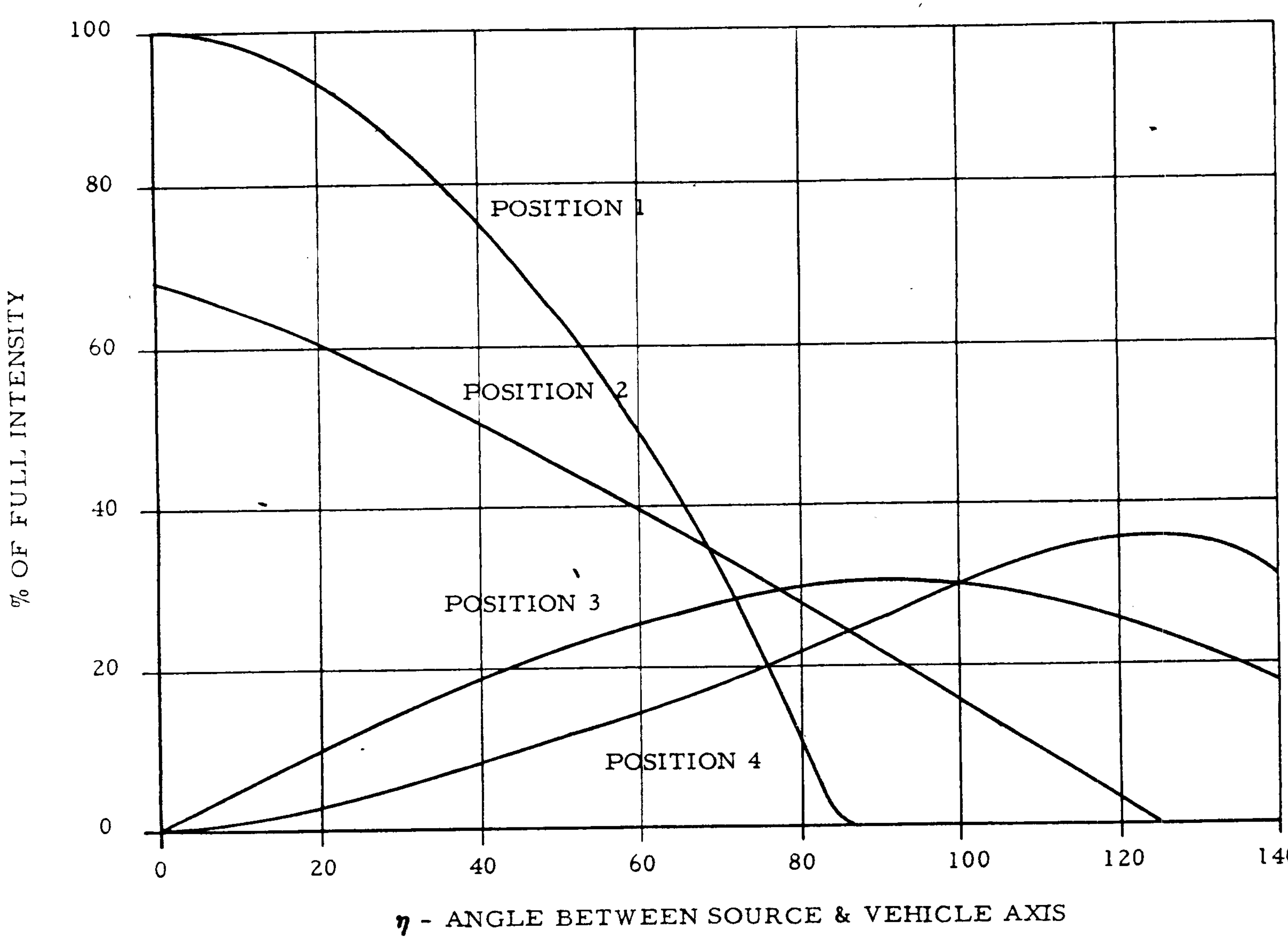


Figure 4

% OF FULL INTENSITY FOR FOUR TORUS POSITIONS - WHITE ENAMEL

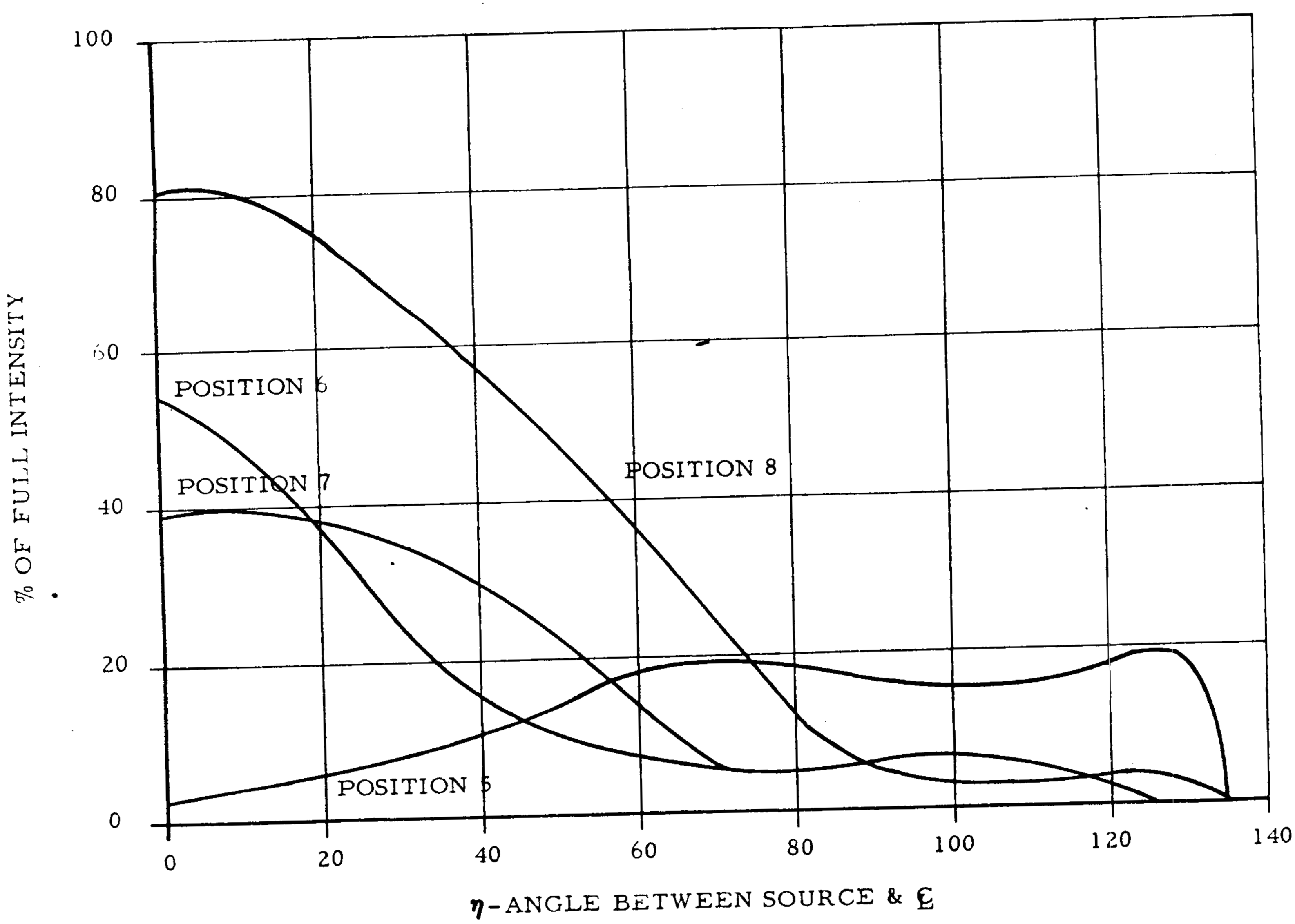


Figure 5

% OF FULL INTENSITY FOR FIVE
SPOKE POSITIONS - WHITE ENAMEL

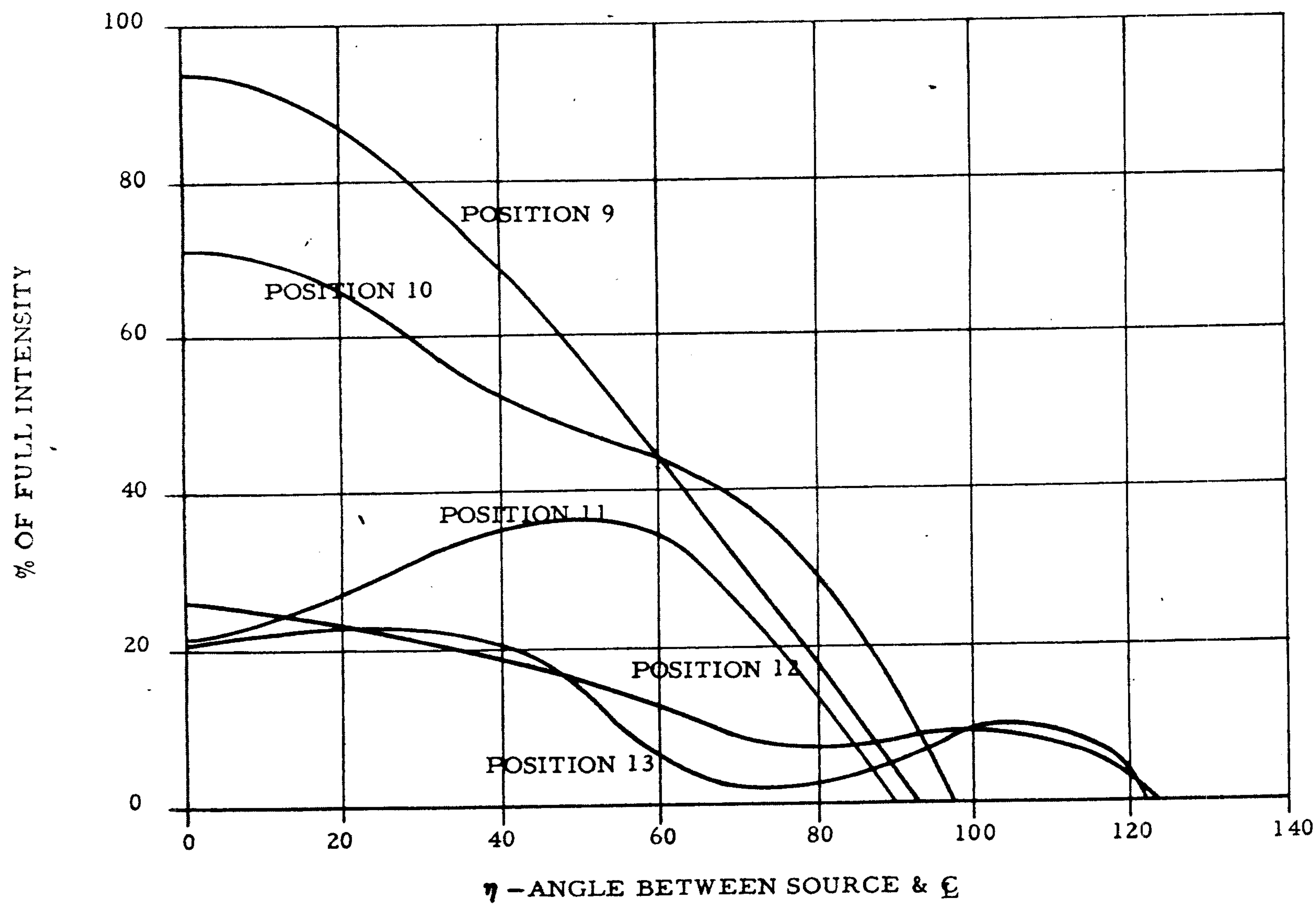


Figure 6

% OF FULL INTENSITY FOR THREE
AREAS IN CENTER - WHITE ENAMEL

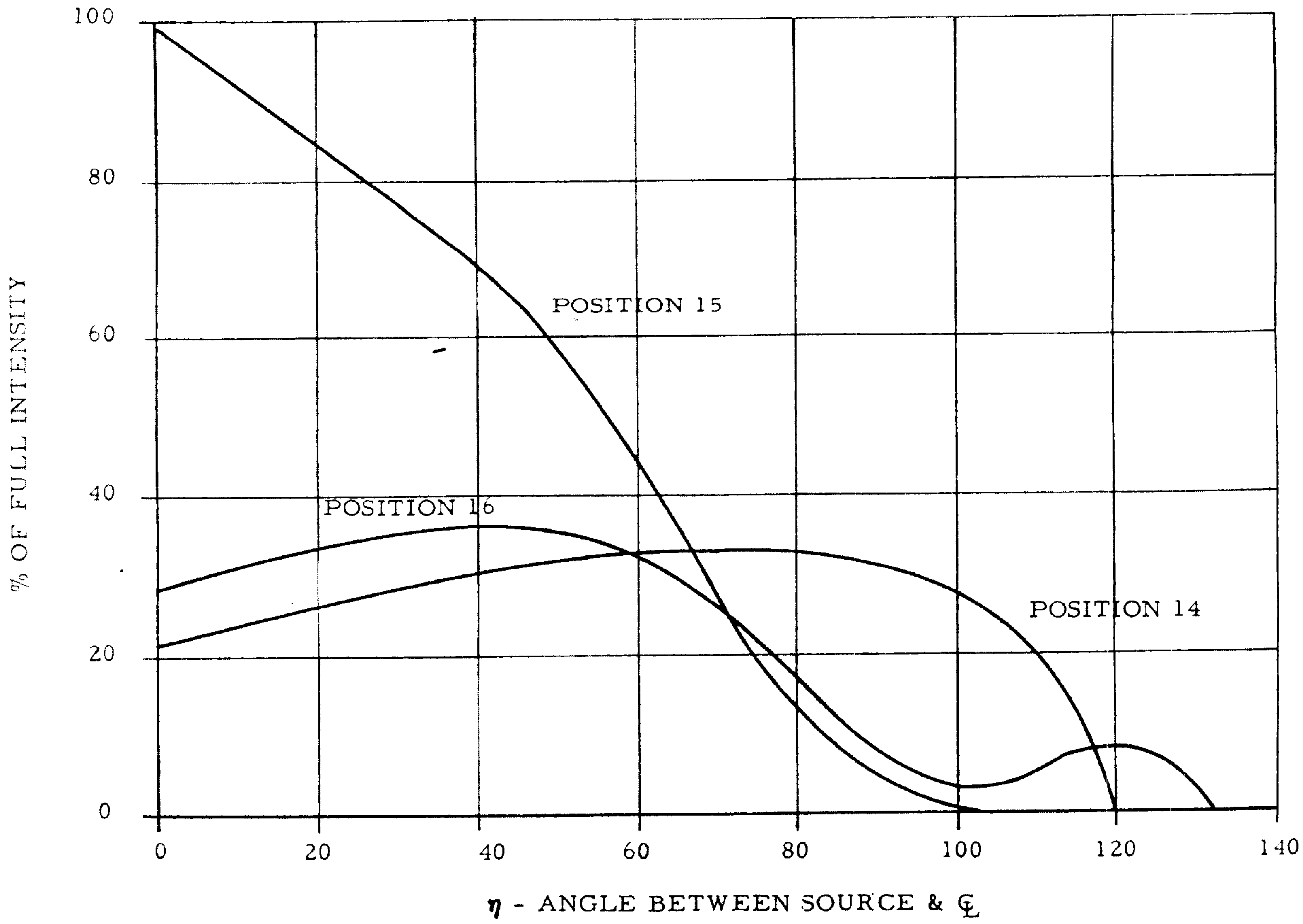
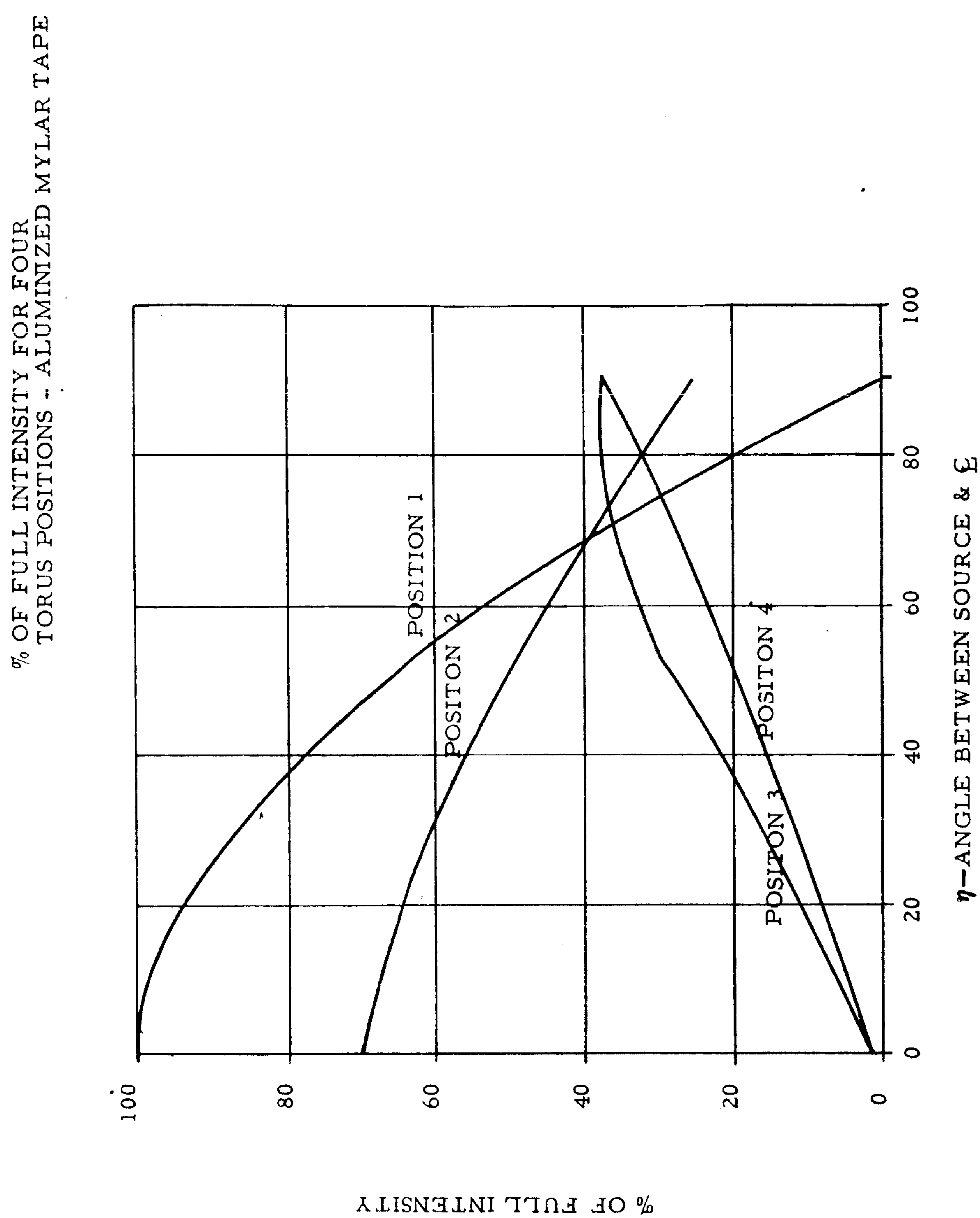


Figure 7



% OF FULL INTENSITY FOR FOUR
TORUS POSITIONS - ALUMINIZED MYLAR TAPE
POSITION 1
POSITON 2
POSITON 3
POSITON 4
100
80
60
40
20
0
% OF FULL INTENSITY
η—ANGLE BETWEEN SOURCE & $\mathcal{E}$

Figure 8

% OF FULL INTENSITY FOR FOUR TORUS POSITIONS - ALUMINIZED MYLAR TAPE

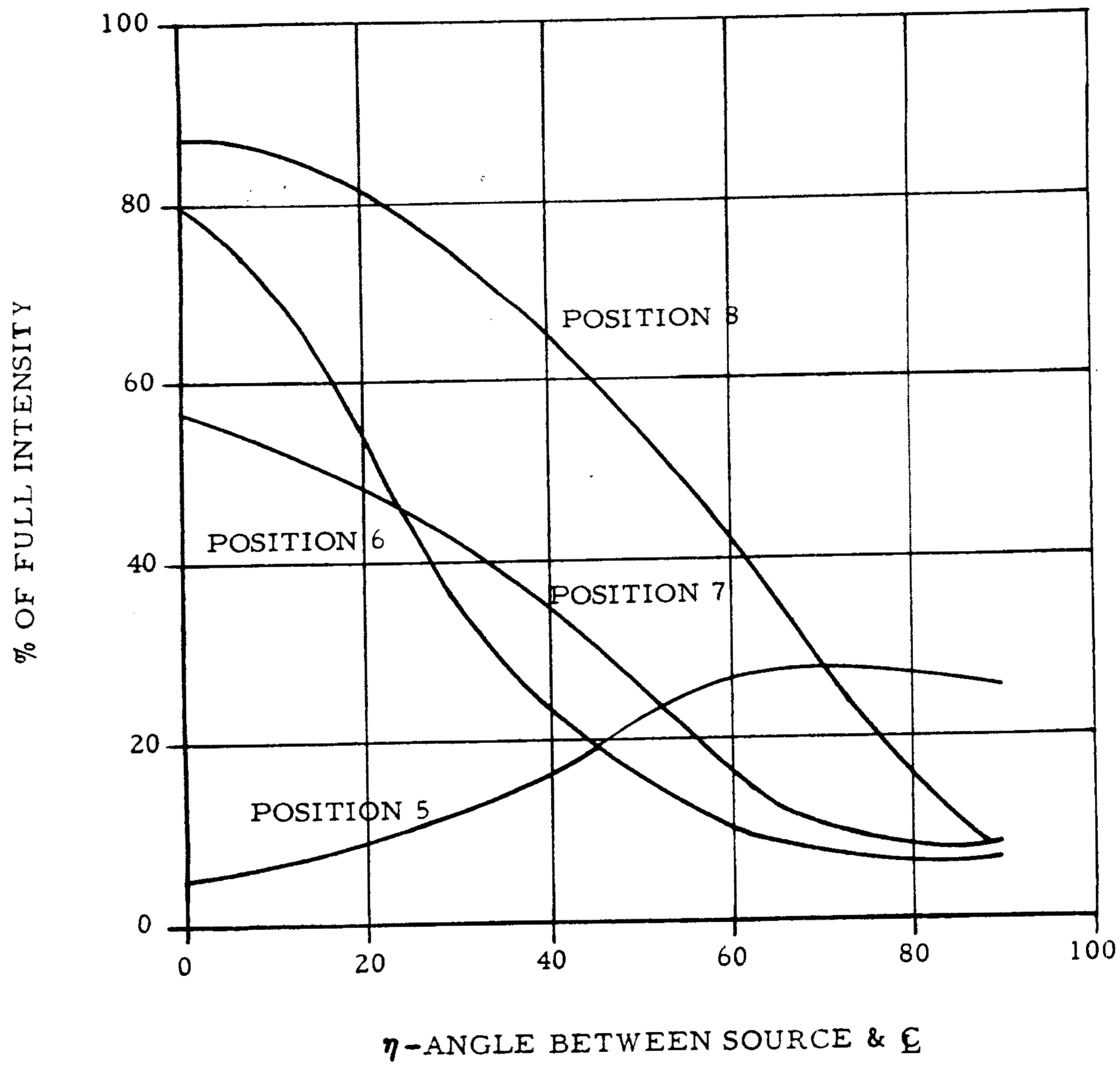
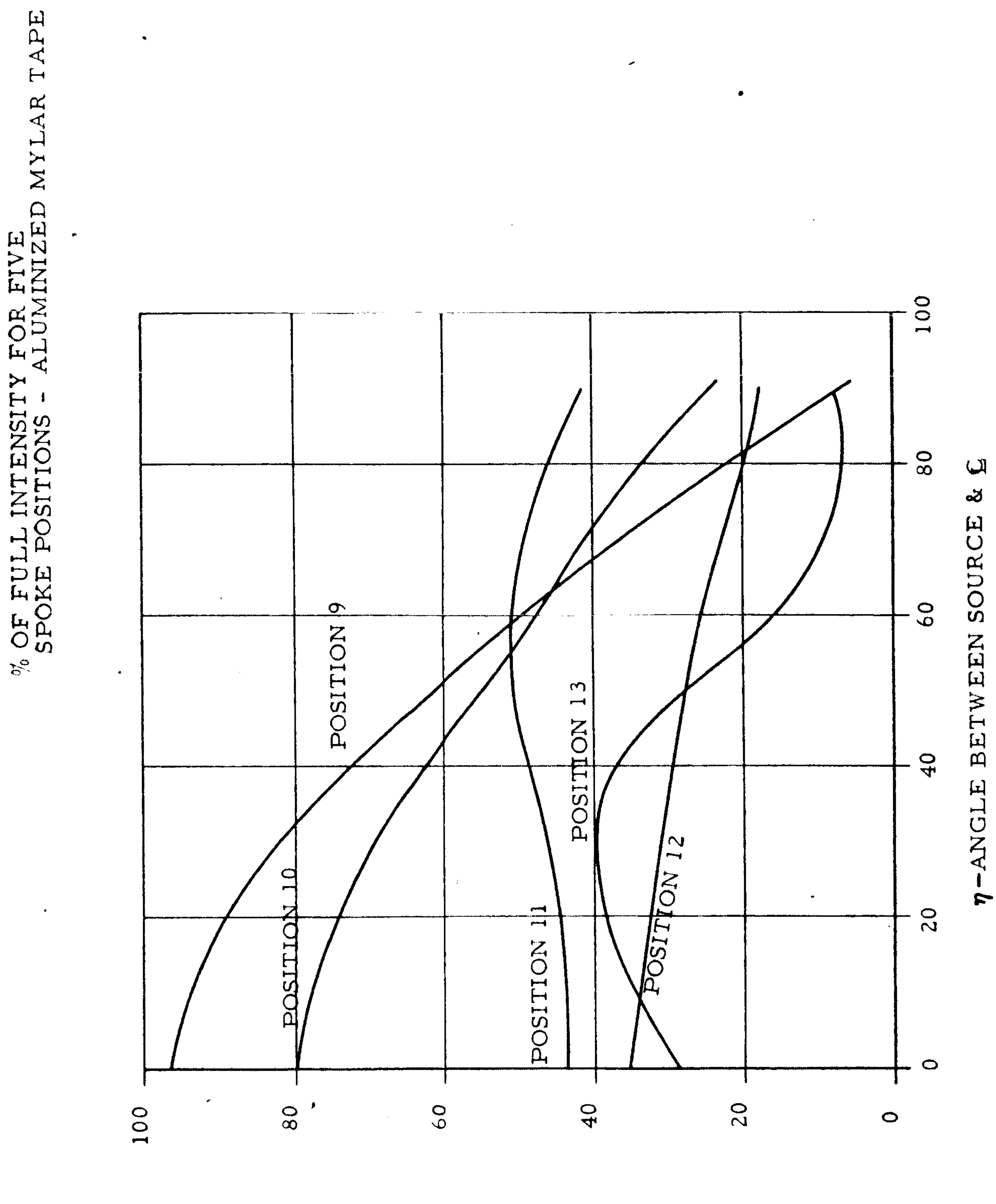


Figure 9



% OF FULL INTENSITY FOR FIVE
SPOKE POSITIONS - ALUMINIZED MYLAR TAPE
POSITION 9
POSITION 10
POSITION 11
POSITION 12
POSITION 13
100
80
60
40
20
0
% OF FULL INTENSITY
η-ANGLE BETWEEN SOURCE & ℄

Figure 10

% OF FULL INTENSITY FOR THREE
AREAS IN CENTER - ALUMINIZED MYLAR TAPE

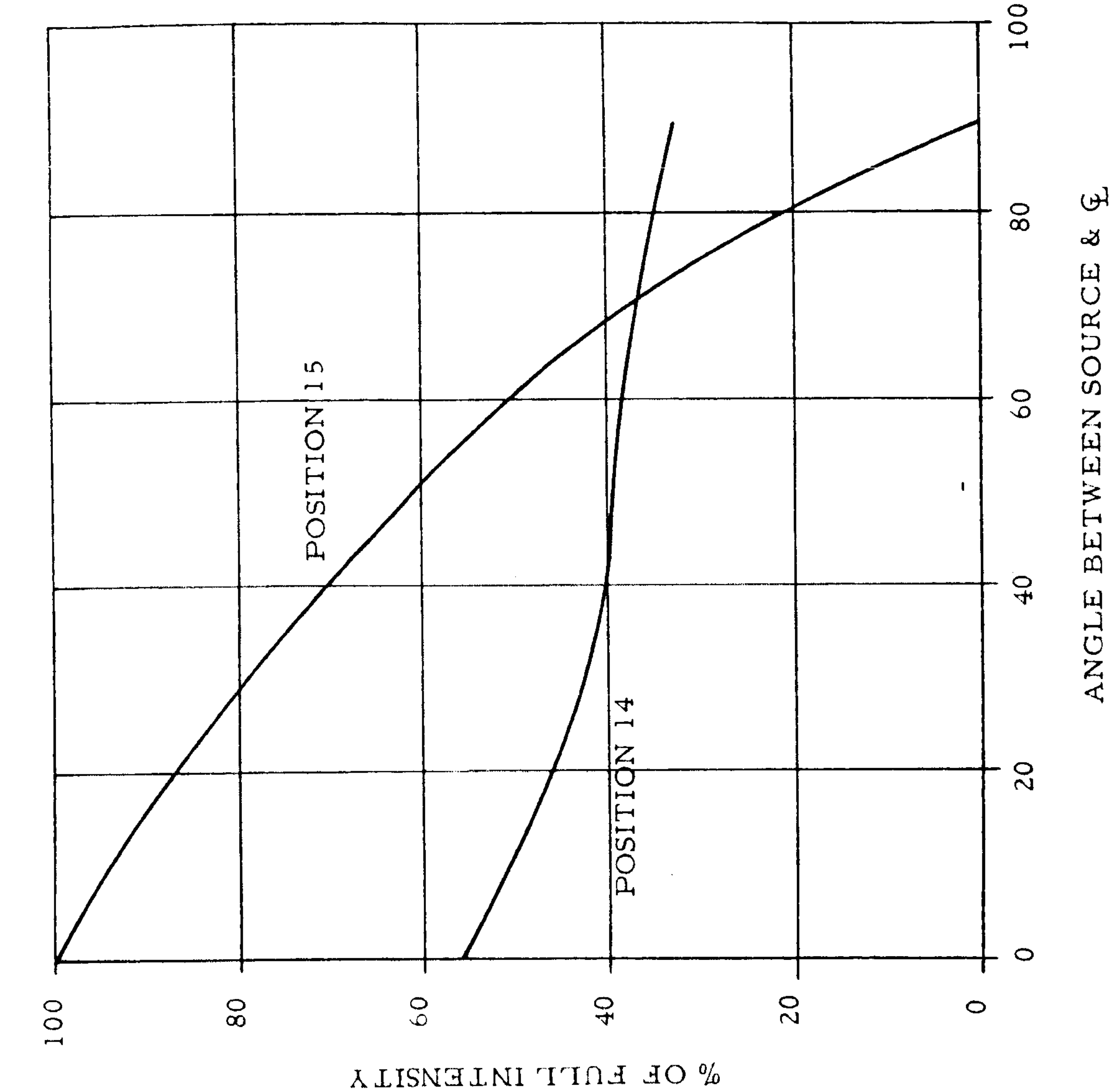


Figure 11

VIEW FACTORS OF SPACE AND EFFECTIVE RADIATING AREA

Section	Actual Area	FS	Effective Area
1	199	.962	191
2	241	.946	228
3	258	.912	235
4	241	.837	202
5	199	.512	102
6	157	.302	47
7	126	.403	51
8	143	.801	115
9	26	.931	24
10	50	.884	44
11	47	.521	24
12	43	.353	15
13	21	.297	6
14	125	.845	106
15	38	.901	34
	Total Effective Area	=	1424 Sq. Ft.

Figure 12

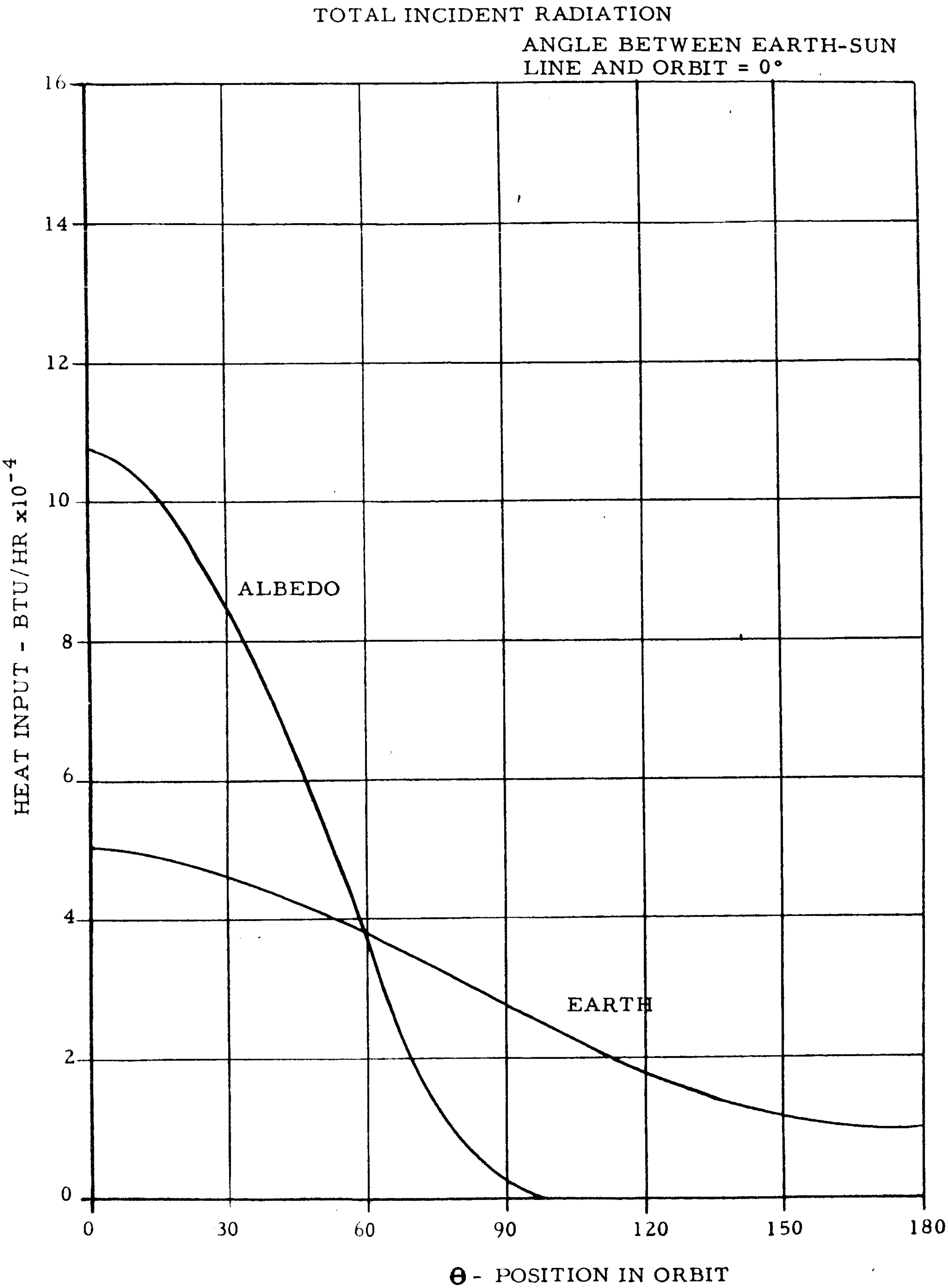

TOTAL INCIDENT RADIATION
ANGLE BETWEEN EARTH-SUN LINE AND ORBIT = 0°
HEAT INPUT - BTU/HR x10^{-4}
16
14
12
10
8
6
4
2
0
ALBEDO
EARTH
0
30
60
90
120
150
180
θ - POSITION IN ORBIT

Figure 13

TOTAL INCIDENT RADIATION

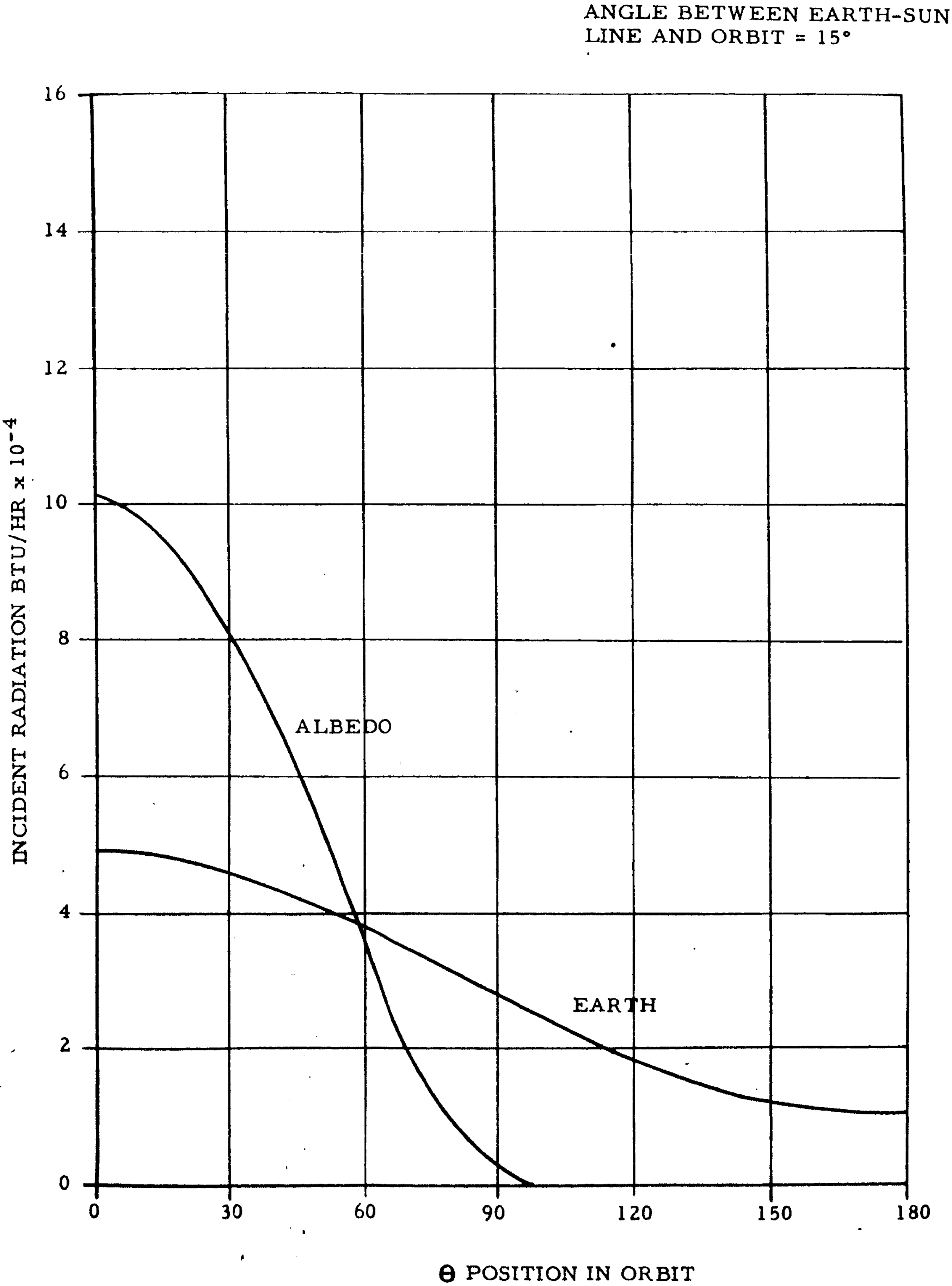


Figure 14

TOTAL INCIDENT RADIATION

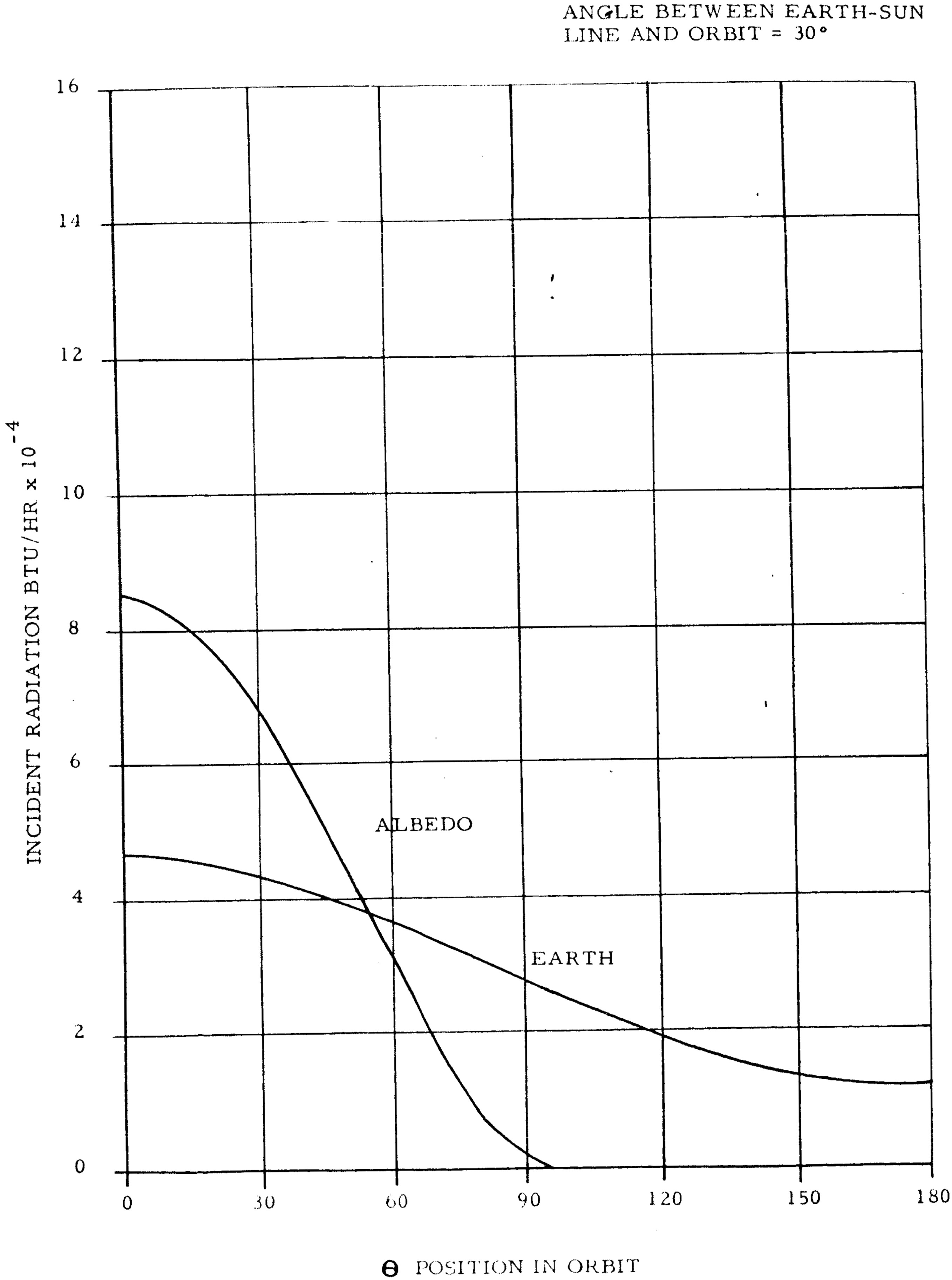


Figure 15

TOTAL INCIDENT RADIATION

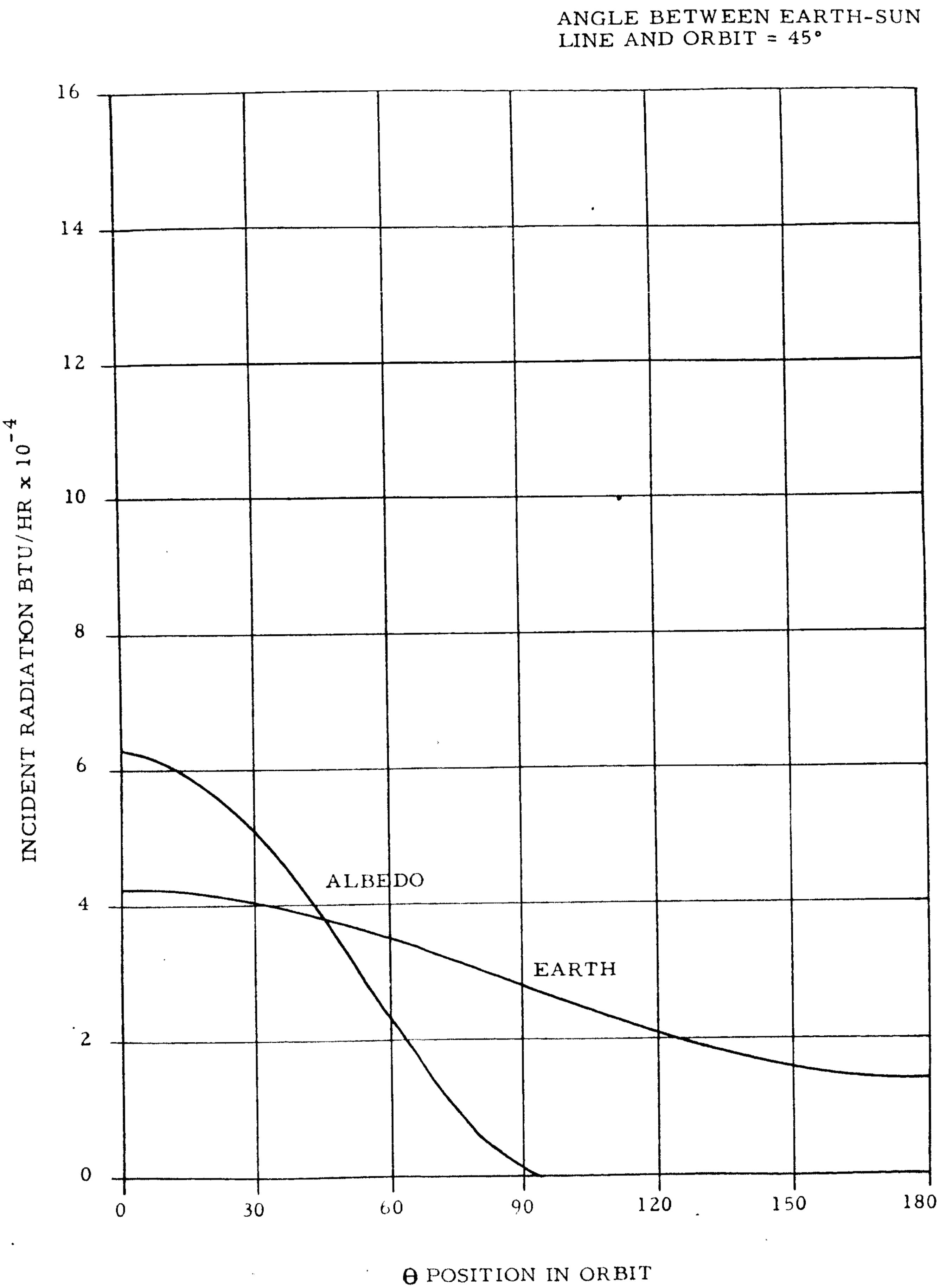


Figure 16

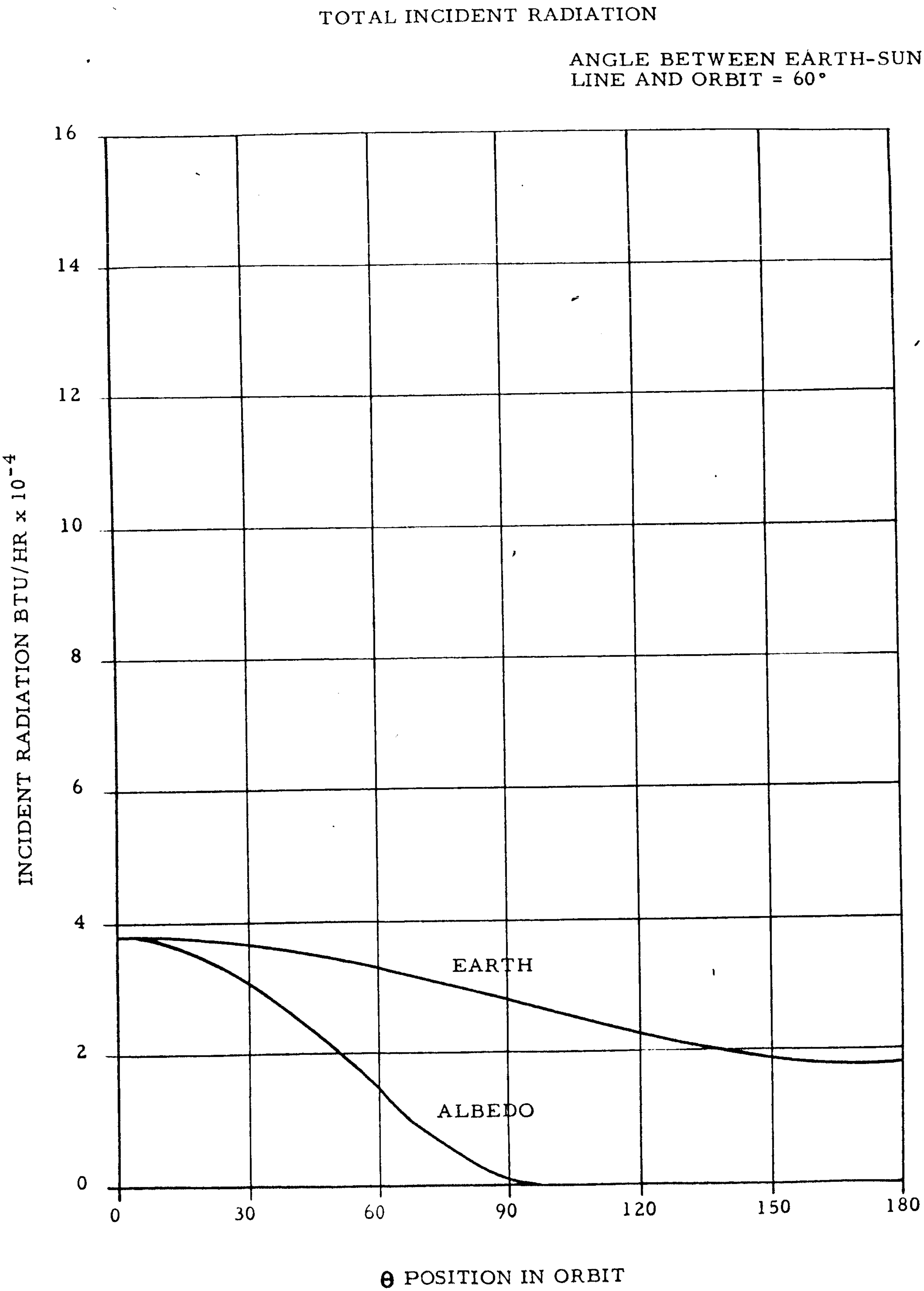

TOTAL INCIDENT RADIATION
ANGLE BETWEEN EARTH-SUN LINE AND ORBIT = 60°
INCIDENT RADIATION BTU/HR x 10^{-4}
16
14
12
10
8
6
4
2
0
EARTH
ALBEDO
0
30
60
90
120
150
180
θ POSITION IN ORBIT

Figure 17

TOTAL INCIDENT RADIATION

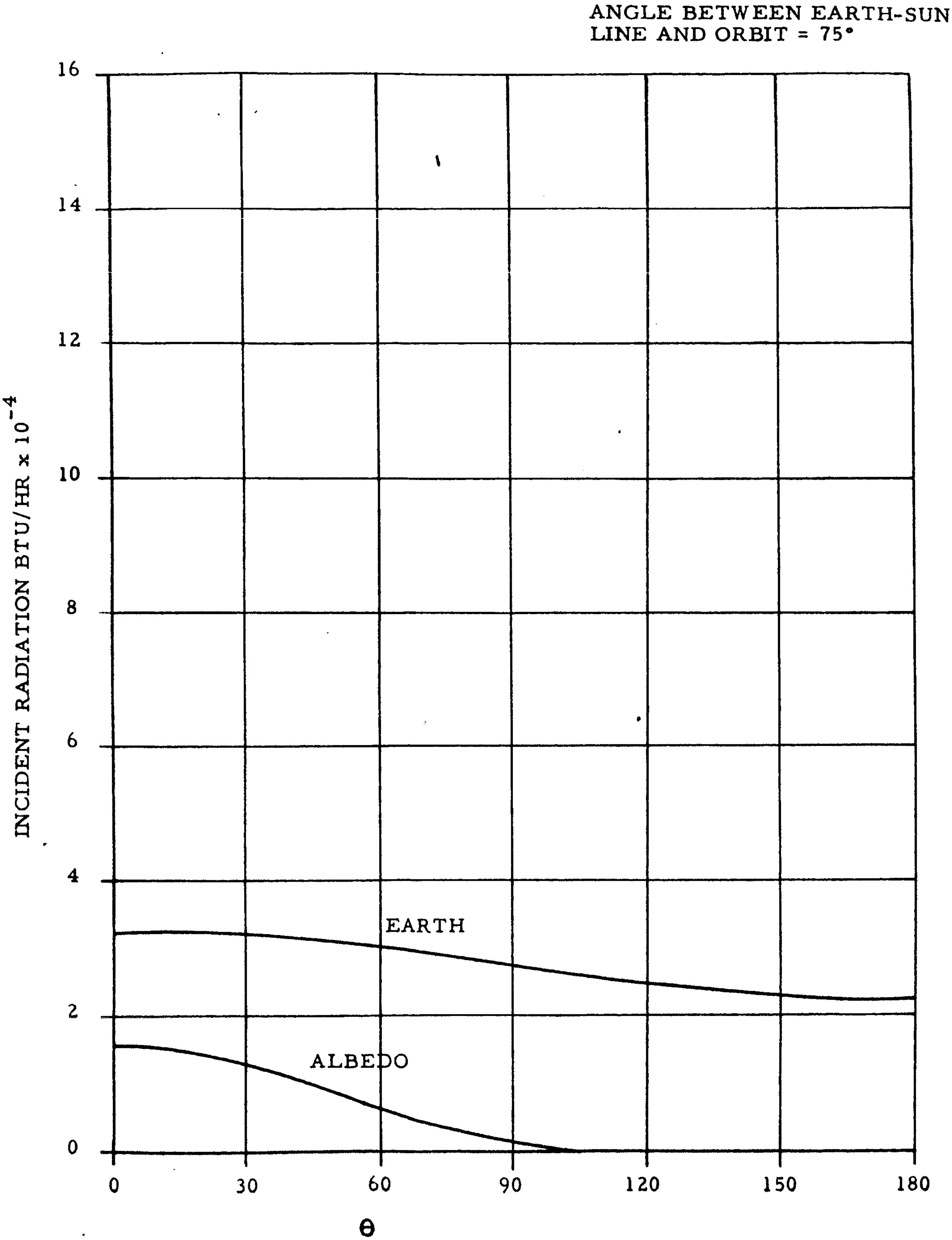


Figure 18

TOTAL INCIDENT RADIATION ON VEHICLE PER ORBIT

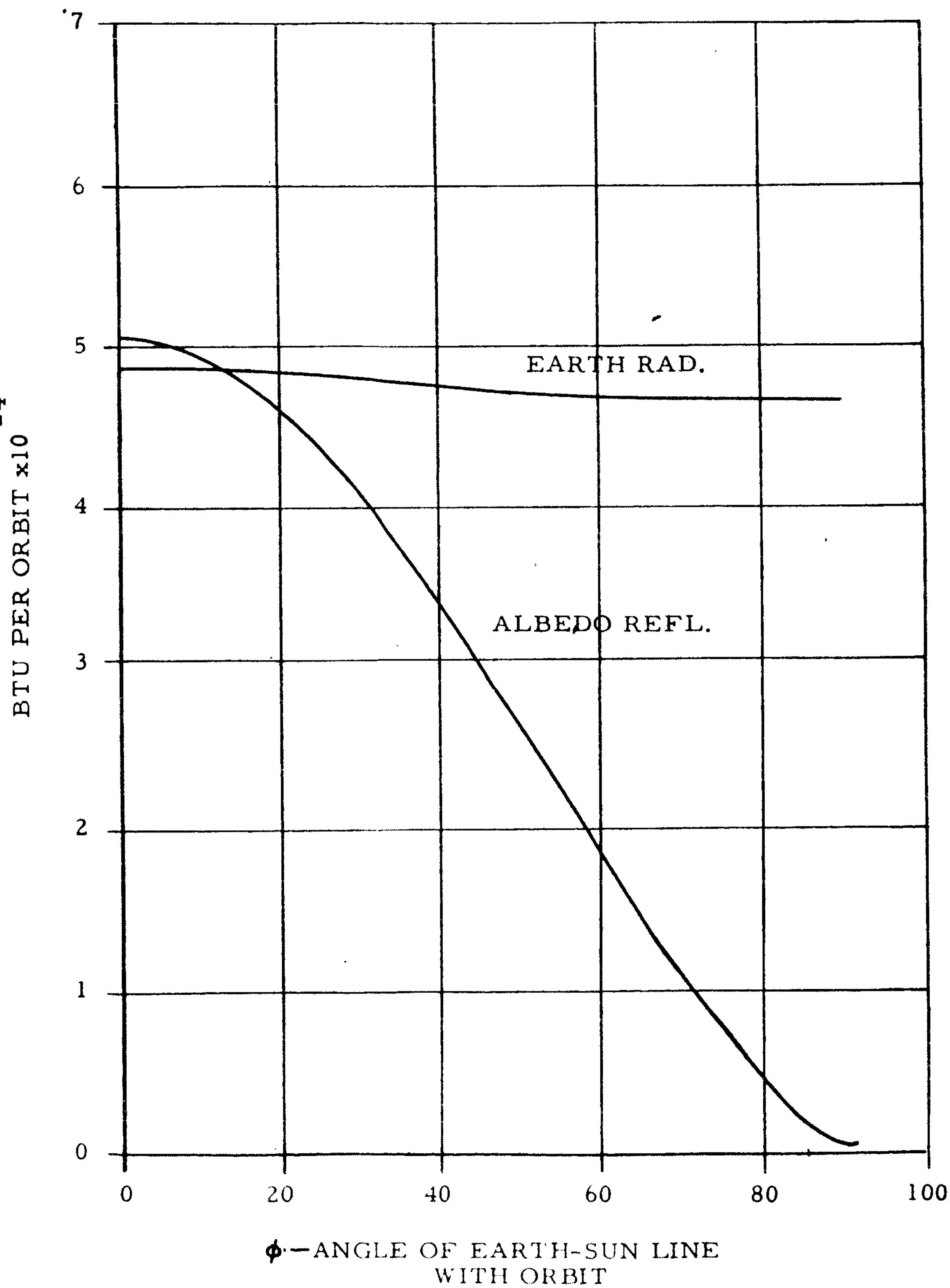


Figure 19

VALUES OF α & ε TO SATISFY TOTAL BALANCE

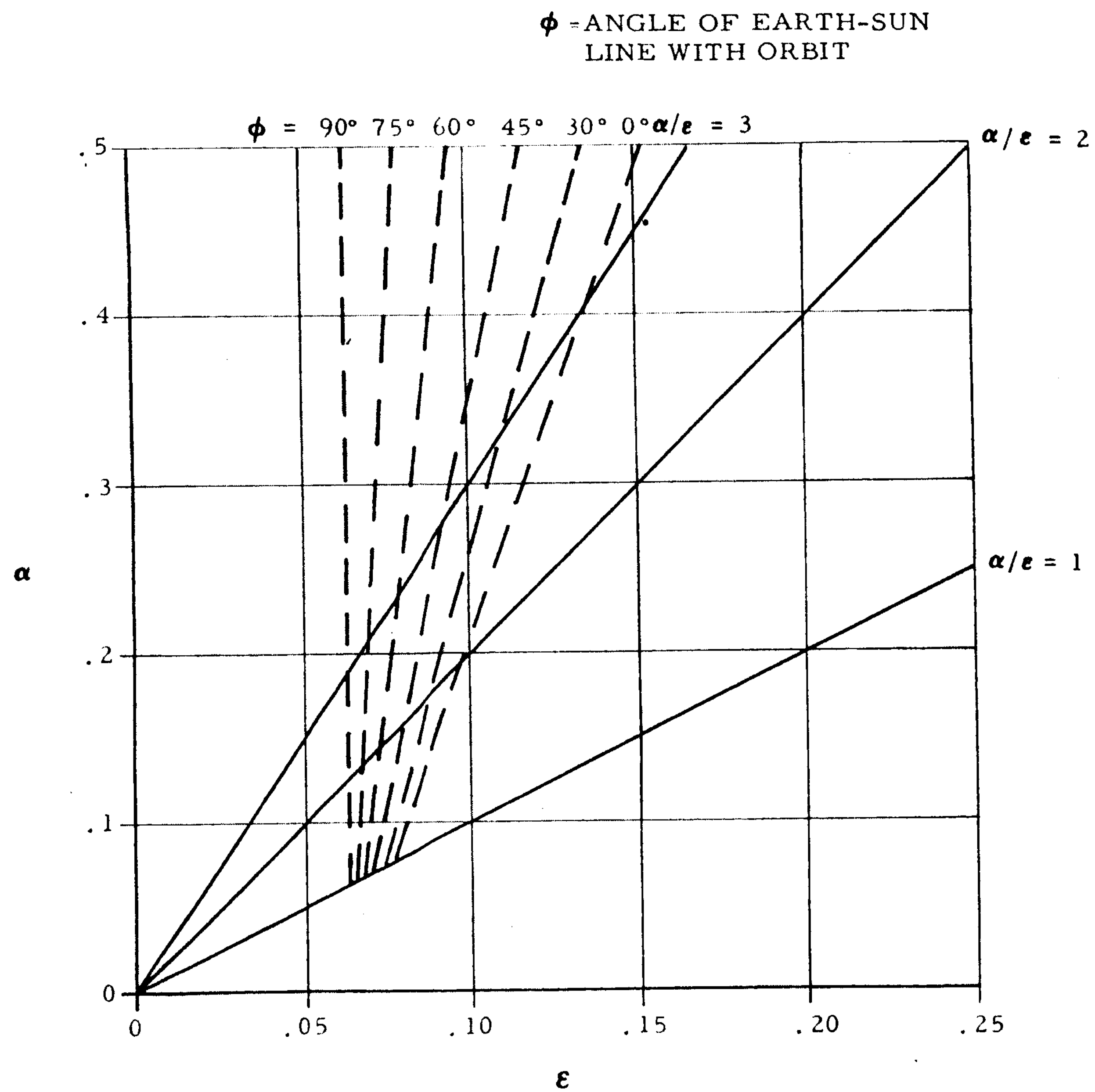


Figure 20

INTERNAL TORUS VIEW FACTORS, F_{ij}

Note: F_{ij} is the fraction of radiation from section i incident on section j.

i \ j	1	2	3	4	5	6	7	8
1	.02	.09	.20	.22	.18	.15	.09	.05
2	.08	.05	.15	.19	.19	.13	.12	.09
3	.15	.13	.08	.13	.15	.13	.10	.13
4	.19	.19	.15	.05	.08	.09	.12	.13
5	.18	.21	.20	.09	.02	.05	.10	.15
6	.17	.22	.22	.16	.06	.01	.04	.12
7	.14	.19	.22	.19	.14	.05	.02	.05
8	.06	.16	.22	.22	.17	.12	.04	.01

INTERNAL SPOKE VIEW FACTORS

i \ j	9	10	11	12	13
9	.0	.15	.24	.29	.14
10	.07	.12	.22	.26	.15
11	.12	.22	.14	.22	.12
12	.15	.26	.22	.12	.07
13	.14	.29	.24	.15	.0

Figure 21

RADIATION IMPINGING ON COLLECTOR

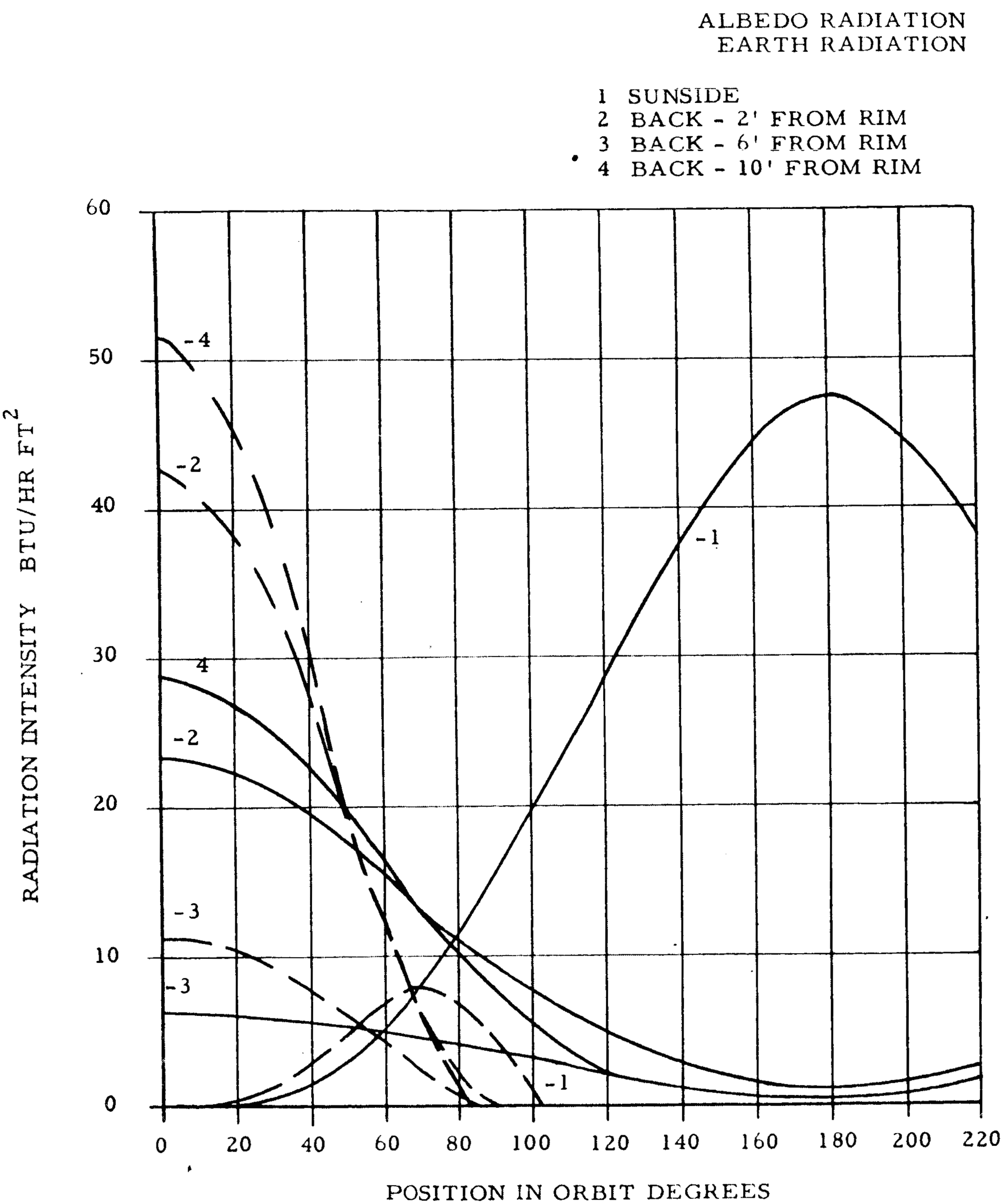


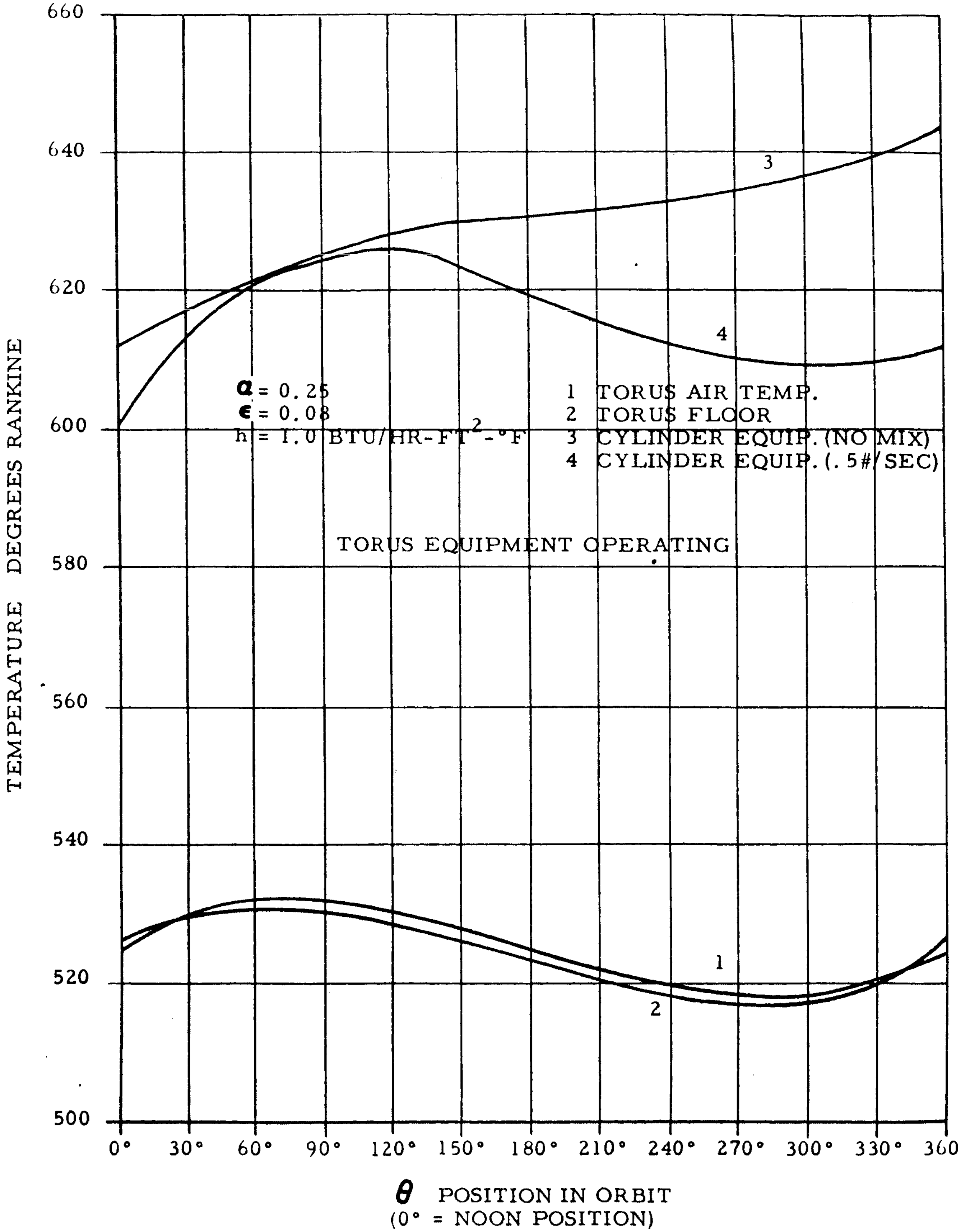
FIG. 22

Sample Temperature for Three Points on the Collector:

A - Two feet from rim
B - Six feet from rim
C - Ten feet from rim

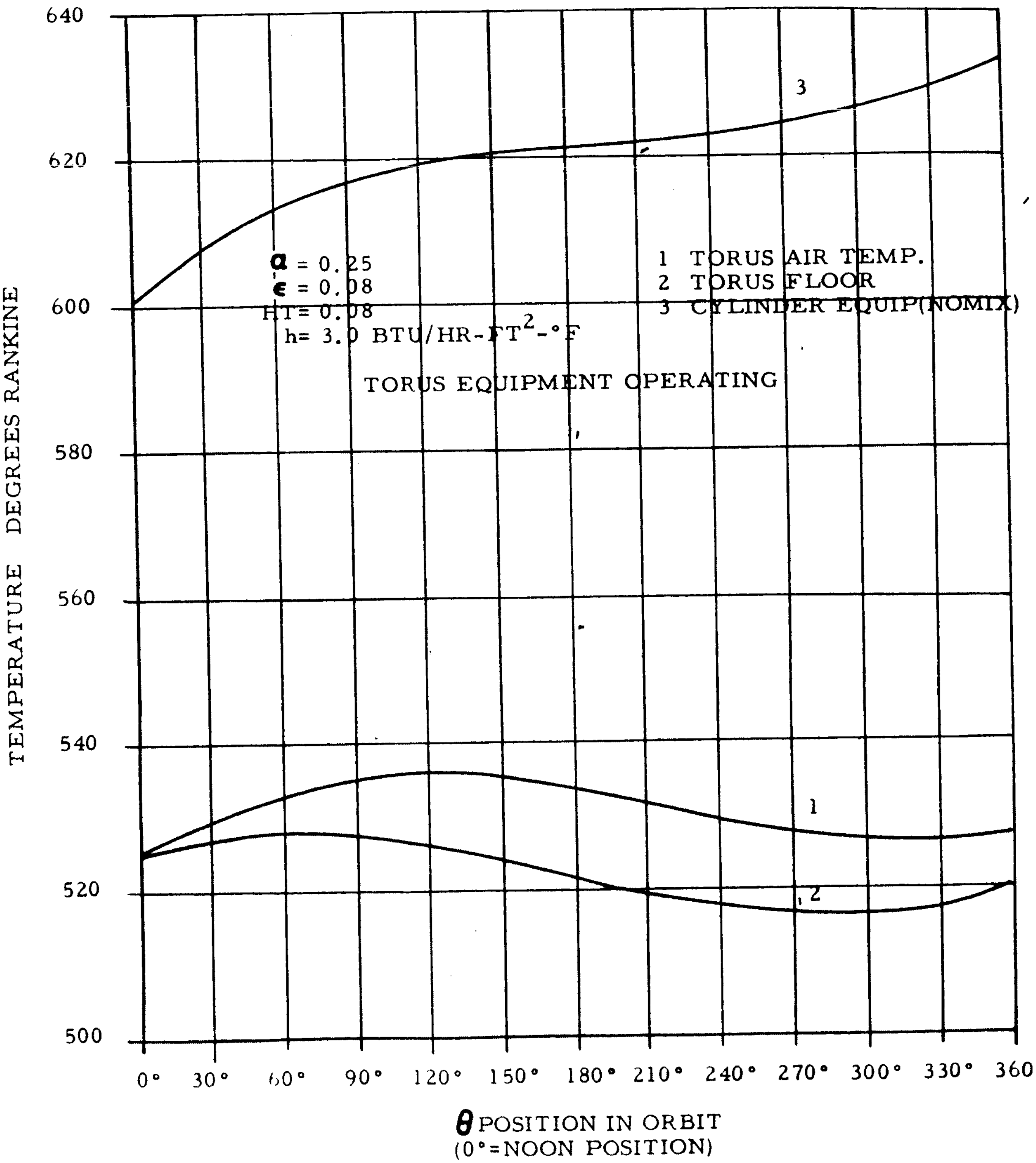
Sun Side		Torus Side			Max. Temp. °R		Min. Temp. °R
α	ϵ	α	ϵ		T_1	T_2	$T_1 = T_2$
.10	.05	.40	.20	A	682	663	356
				B	694	676	436
				C	706	689	375
		.40	.40	A	621	600	334
				B	651	631	420
				C	650	629	354
		.40	.80	A	564	540	312
				B	603	581	397
				C	596	573	330
		.60	.60	A	601	579	321
				B	629	608	407
				C	634	613	340
		.80	.80	A	589	566	312
				B	613	591	397
				C	624	602	330
.60	.60	.40	.40	A	672	642	382
				B	682	658	411
				C	680	654	389
		.40	.80	A	647	606	367
				B	668	635	403
				C	659	623	375
		.60	.60	A	662	627	374
				B	676	648	406
				C	673	633	381
		.60	.80	A	651	611	367
				B	669	637	403
				C	664	629	375
		.80	.80	A	655	617	367
				B	670	639	403
				C	668	636	375

Figure 23



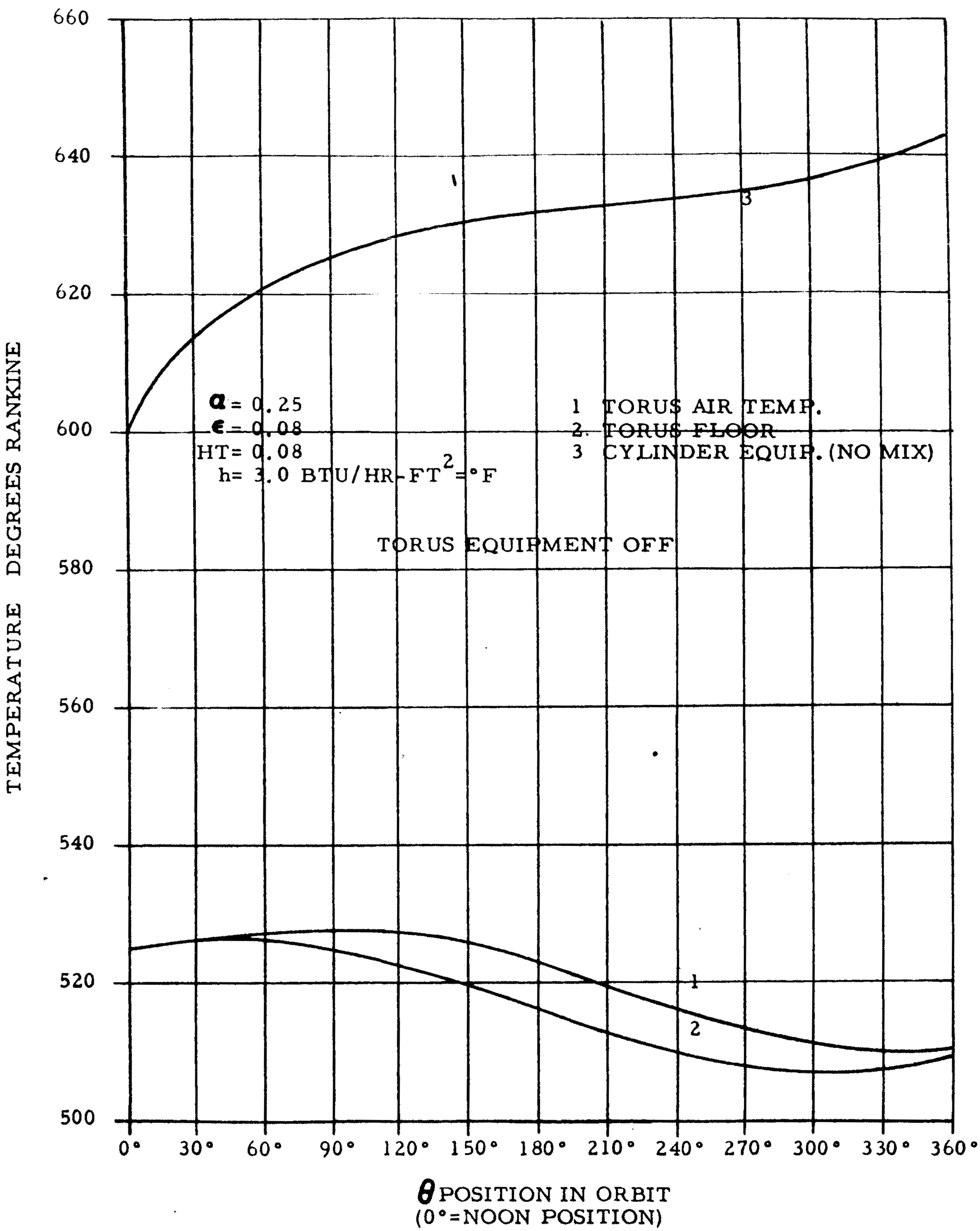
VEHICLE TEMPERATURES DURING ORBIT

FIG. 24



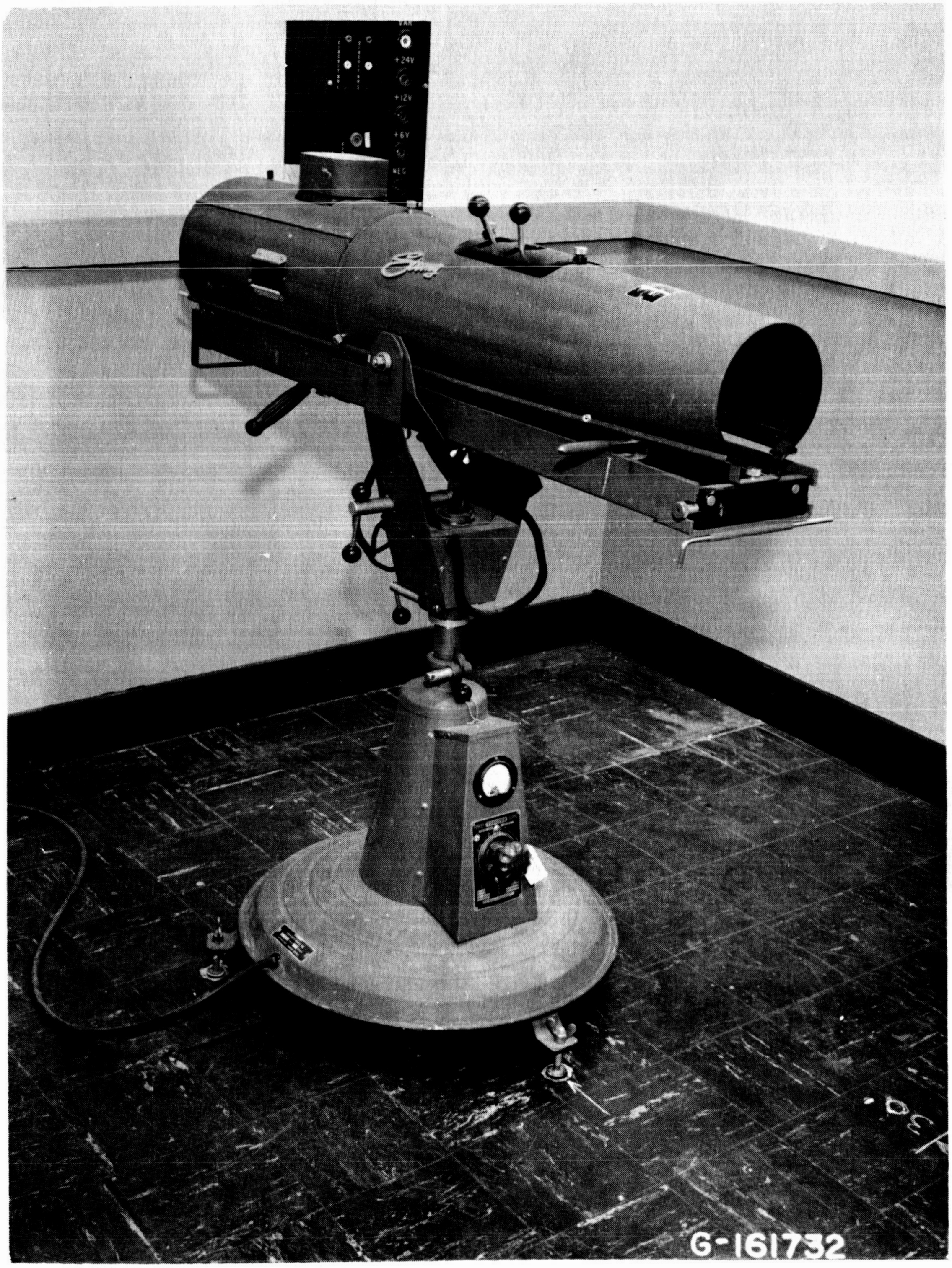
VEHICLE TEMPERATURES DURING ORBIT

FIG. 25



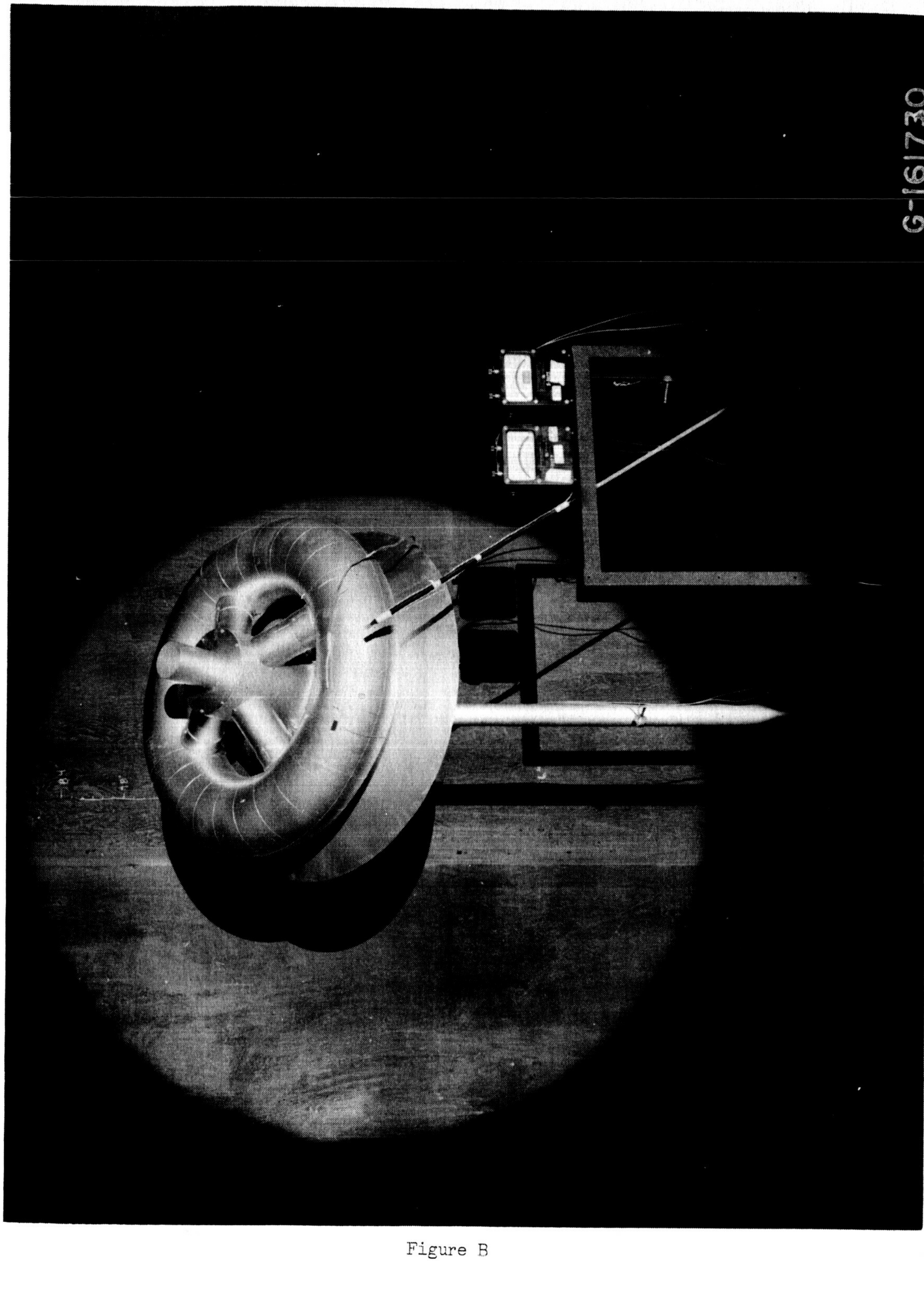
VEHICLE TEMPERATURES DURING ORBIT

FIG. 26



Carbon-Arc Lamp-Hi-Intensity Collimated Source

Figure A



Reflection Test Model with Diffuse Surface

Figure B

Reflection Test Model with Specular Surface

Figure C